DOCUMENT NO. 63SD801

15 OCTOBER 1963

356p. 23 refs

N65-35279 (ACCESSION NUMBER) (THRU)

356 (PAGES) 1 (CODE)

CR 51835 (NASA CR OR TMX OR AD NUMBER) 31 (CATEGORY)

FACILITY FORM 602

GPO PRICE $ ______

CFSTI PRICE(S) $ ______

Hard copy (HC) 7.00

Microfiche (MF) 2.00

ff 653 July 65

VOYAGER DESIGN STUDY,

VOLUME II:

MISSION AND SYSTEM ANALYSIS

(NASA

~~Prepared Under~~ Contract NAS W-696)

for

(NASA CR-51835; Rept. 63SD801)

NATIONAL AERONAUTICS AND SPACE ADMINISTRATION

OFFICE OF SPACE SCIENCES

WASHINGTON, D.C.

[redacted]

GENERAL ELECTRIC, Philadelphia, Pa.

MISSILE AND SPACE DIVISION

Valley Forge Space Technology Center

P.O. Box 8555 • Philadelphia 1, Penna.

VOLUME SUMMARY

The Voyager Design Study report is contained in six volumes, an appendix, and subcontractor reports. The volume numbers and their titles are as follows:

Volume No.	
I	Voyager Design Summary
II	Mission and System Analyses
	1. Mission Analysis 2. Parametric System Performance 3. Voyager Systems 4. Reliability
III	Subsystem Design
	1. Communications 2. Television 3. Radar 4. Guidance and Control 5. Propulsion 6. Power Supply Appendix (Classified)
IV	System Design
	1. Entry/Lander 2. Orbiter
V	Sterilization
VI	Program Development Plans

Separate Reports from the following Companies are also included:

Aerojet-General Corp.
Barnes Engineering
Bell Aerosystems Co.
Conductron Corp.
Electro-Mechanical Research Inc.
General Electric Co.
  Light Military Electronics Dept.
General Precision Inc.
Hazeltine
North American Aviation Inc.
  Autonetics Division
  Rocketdyne Division
Radio Corporation of America
Rocket Research Corp.
Texas Instruments Corp.
Thiokol Chemical Corp.
  Elkton Division
  Reaction Motors Division

[redacted]

[redacted]

VOLUME II

CONTENTS

CONTENTS (Continued)

CONTENTS (Continued)

ILLUSTRATIONS

ILLUSTRATIONS (Continued)

ILLUSTRATIONS (Continued)

TABLES

TABLES (Continued)

SECTION NO. 1 MISSION ANALYSIS

1.1 SCIENTIFIC OBJECTIVES

The scientific objectives of the Voyager program, as enunciated by NASA are listed below in order of decreasing priority.

PRIMARY OBJECTIVES

1. Detection and characterization of life
2. Exploration of planetary surfaces and interiors
3. Investigation of planetary atmospheres

The detection of life on other planets of the solar system would have a tremendous impact on our ideas of Man's place in the scheme of things. It should obviously be of first priority in the Voyager program. A detection of life, however, would whet the appetite for a full characterization of that life. Such a characterization is within the scope of Voyager capabilities and should be made an integral part of the primary objective. As we know the planets at present, Mars appears to be the most likely abode of life, a fact which conditions the preliminary planning phases of Voyager. However, one should not overlook the possibility that life has developed in the more hospitable parts of Venus also.

Knowledge of the properties of the planets themselves, including both the solid portions and the atmospheres, is of a great deal of interest for obvious reasons. Atmospheric measurements have been given a somewhat lower priority for Voyager than that assigned to surface investigations because it is anticipated that atmospheric measurements will be performed by other vehicles such as Mariner "B" capsules before the Voyager mission arrives on the scene.

SECONDARY OBJECTIVES

Secondary, but still important, objectives for Voyager are

1. Acquisition of data on interplanetary environments, and
2. Development of concepts and acquisition of information necessary for the development of manned exploration systems.

The first of these will be accomplished mainly by vehicles which precede those of Voyager, but additional data will be valuable in determining time and space variability of the parameters. Thus, in-transit measurements will be included where they do not degrade the other measurements appreciably. The point at which such degradation becomes "appreciable" must be a matter of judgement.

Although it might appear from the above that the eventual manned exploration of Mars is of minor concern to Voyager, such is by no means the case. The applicability of many of the measurements to exploration by man is direct indeed, and it is envisioned that Voyager will serve as the foundation on which the necessary planning will be based. Surface material properties and atmospheric characteristics are of particular note in this respect. An additional objective, however, is to go even further in Voyager and show the way to a smooth evolutionary expansion by which the manned exploration can be accomplished.

1.2 MISSION CONSTRAINTS

There are a number of constraints within which one must work, some of the constraints being natural in nature, some being imposed by practical engineering considerations, and a very important one being wisely and arbitrarily imposed by NASA to prevent premature defiling of the planets themselves. A few of the more important constraints are the following:

1.2.1 OPPORTUNITIES FOR MARS AND VENUS

The relative positions of the planets with respect to that of the Earth are optimum for launching planetary probes during only relatively restricted time periods - the firing "windows." The windows are nominally about thirty days in length and are repeated at intervals of about 25 months for Mars and 19 months for Venus. These facts are of practical importance in that if the launching is not accomplished during the short time period one must wait one and a half or two years before another chance presents itself. The extreme complexity of both the launching procedure and the spacecraft itself makes necessary a very thorough overall mission plan to assure a timely launch of the mission.

A very important fact which must be considered in planning the long-term exploration is that the propulsion energy requirements varies from one opportunity to another. The opportunities presently under study have relative rankings as follows:

Mars 1969 - moderate
Mars 1971 - very good
Mars 1973 - good
Mars 1975 - poor
Venus 1970 - moderate
Venus 1972 - moderate

It is seen that emphasis should be put on the 1969, 1971 and 1973 Mars missions to obtain many data from those favorable opportunities. The timing is less critical for Venus since the orbit of Venus has a smaller eccentricity resulting in less variation of energy requirements among the mission opportunities.

1.2.2 TYPES OF TRAJECTORIES AND ARRIVAL TIMES

The type of trajectory chosen for the spacecraft is dictated mainly by engineering considerations, but it is of importance from the scientific standpoint through its effect on arrival time at the planet. Results of the present study are summarized in Table 1.2.2-1 and the arrival times for the Mars missions are shown schematically in Figure 1.2.2.-1.

TABLE 1.2.2.-1 TRAJECTORIES AND ARRIVAL TIMES FOR THE VARIOUS PLANETARY MISSIONS

	Mars				Venus	
	1969	1971	1973	1975	1970	1972
Type of trajectory	II	I	I	II	I	II
Transit time (days)	270-280	210-225	205-220	308-322	98-121	163-190
Arrival at planet N. Hemisphere season	Late fall to early winter (325^o-15^o)	Early winter (0^o-52^o)	Early spring (90^o-110^o)	Mid-spring (133^o-145^o)	-	-

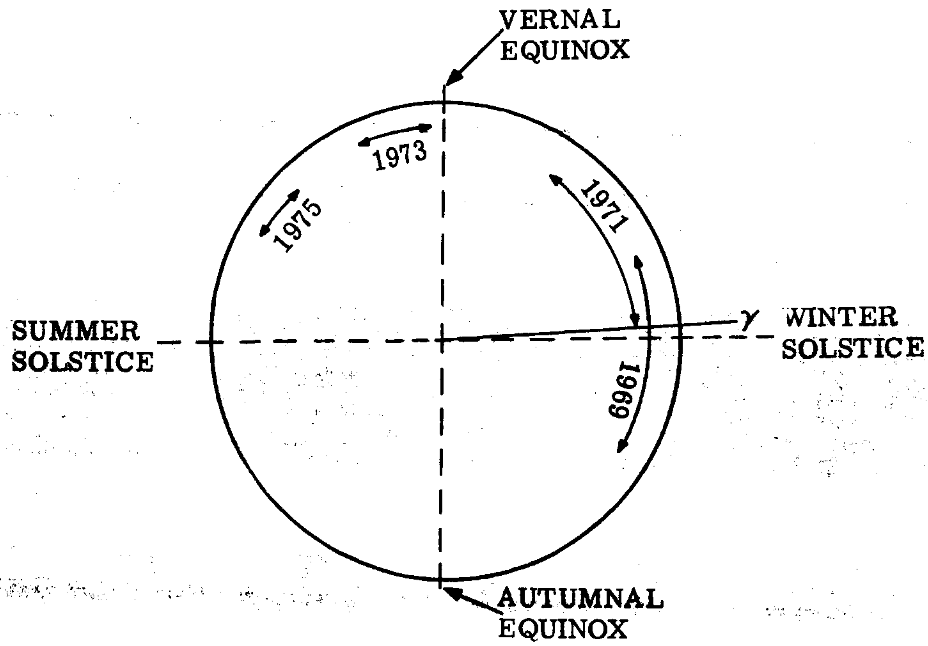


Figure 1.2.2-1. Approximate Northern-Hemisphere Seasons of Mars at Which the Spacecraft of the Various Missions Will Arrive at the Planet

Since the seasonal variations for Venus are still unknown, the equivalent diagram for Venus missions must await more adequate data on the rotational characteristics of the planet.

1.2.3 UTILIZATION OF PRIOR DATA

More in the nature of an asset than a constraint in the Voyager planning is the specification that atmospheric data obtained in the Mariner program will be available prior to any Voyager flights. These data will be of value in determining restraints on entry corridors, evaluating the priority assigned to atmospheric measurements; and for the 1971 opportunity and beyond, permit a more optimum entry vehicle design.

1.2.4 CONTAMINATION OF PLANETS

A firm ground rule has been imposed by NASA that every feasible precaution be taken against introducing living organisms into the planets before it is definitely established whether or not life exists there already and, if so, what its characteristics are. Vehicles which enter the atmosphere must be sterilized to the extent that the probability of introducing a living organism is less than one in ten thousand. Vehicles which orbit the planet must be either similarly sterilized if the probability of impacting is greater than one in ten thousand or be assured of a minimum time of fifty years in orbit before impacting the planet. As a result of some work at JPL, the latter criterion is satisfied by setting a minimum altitude of 1800 km for circular orbits and a minimum altitude of 1500 km for elliptical orbits.

The effect of the sterilization requirement on the mission is a serious one. Some materials are degraded by the high temperatures required, namely, a minimum of 135°C for a minimum of 24 hours. Biological assays seem to be particularly vulnerable to high temperature. Chemical sterilization procedures likewise have serious adverse effects on biological experiments. The situation is sufficiently serious that it appears likely that some otherwise attractive biological experiments may be eliminated because of the sterilization requirement. Atmospheric and surface experiments are much more tolerant of the sterilization procedures, although it must be emphasized that further work is needed in the area. The effects of the sterilization requirements on both experiments and engineering subsystems are discussed in detail in Volume V of this report.

Whether or not radioactive material constitutes a "contaminant" to the planets has not been defined, but it is a question which should be considered. It is anticipated that the power supply for Mars landing vehicles may contain curium 244 which is radioactive. Since measurements of the radioactivity of the surface material of the planet are important scientifically, it is undesirable that the natural radioactivity be modified to an appreciable extent by that which is introduced, especially before adequate measurements of the natural conditions are obtained.

The problem of radioactive contamination is different, however, from that of biological contamination. One is not merely detecting the presence of radioactivity as one is detecting the presence of life. In all probability, there are radioactive materials on Mars just as there are on the Earth. In fact, the thinner atmosphere of Mars will permit a stronger cosmic ray bombardment of the surface material than that experienced by the Earth's surface, a condition which would produce greater radioactivity on Mars than on Earth. Consequently, the introduction of a small amount of radioactive material would cause a measurable change of total activity only in the immediate vicinity of the source itself and would not constitute a serious contamination of the planet. Even dispersal of the material over wider areas, such as might result from breakup of the entry vehicle, would not appreciably affect the natural total radioactive background, although of course the abundances of curium relative to other trace elements might be altered significantly by such a dispersal.

1.3 CRITERIA FOR MISSION SUCCESS

In order to perform a meaningful reliability evaluation one must establish some criteria of mission success. The probability of complete and perfect performance of all vehicles, systems, and instruments is vanishingly small for such a complicated mission. What, then, is a realistic goal against which the success or failure of a mission may be gauged?

This judgement must take into account at least 1) total data yield, 2) relative values of the different data types, and 3) data value versus time. In addition, a relative weighting for the different possible missions is very desirable at the outset, since a group of data which would be thought of as constituting a success for say Venus 1970 might not be similarly considered for a Mars 1975 mission.

The one mission for which detailed success criteria have been established in the present study is that of Mars 1969. The goal for operable lifetime of the Mars 1969 orbiter is three months, and that for each of the landers is six months since beyond these times the mission values and probability of success decrease significantly. The limiting item by which orbiter life is gauged is the amount of gas supplied for vehicle stabilization. Without stabilization the vehicle is incapable of adequate communications or taking useful measurements. The possible life of the Landers is longer. Since they will be equipped with radioactive thermoelectric power supplies, there is a possibility of several years lifetime capability for operation of the experiments and the low-data-rate communication system on each of the Landers. A goal of a minimum of six months operation has been established, although longer periods are desirable and may be attained.

For the purposes of the present study, a mission "success" has been defined as providing a total data yield of 75 percent of that possible within the three-months Orbiter life and six-months Lander life. This is obviously an arbitrary judgement and is subject to review, but it seems a reasonable value for present purposes. It has been used in the reliability analysis described in Section 4.0 of this volume.

It is evident that not all measurements are of equal value in assessing the total worth of the mission. Since the major objective of the 1969 Mars mission is that of life detection, biological experiments should be emphasized in determining success or failure. Conversely, atmospheric measurements are downgraded in that mission because of the assumption that atmospheric measurements will have been performed on the Mariner project prior to 1969.

The apportionment of mission value which has been assigned to the experiments suggested for the Mars 1969 mission is indicated below. The various parts of the mission were arbitrarily given the following value point assignments:

Each Lander during entry	10 points
Each Lander on surface	60 points
Orbiter	30 points

The emphasis on Landers reflects the need for surface measurements in the unambiguous detection of life on Mars, although the orbital measurements (principally television pictures) are not unimportant in its characterization.

The detailed breakdown of points assigned to the various experiments is shown in Table 1.3-1. The points were derived on judgement of scientific value alone, but it is logical that the efficiency of each experiment in meeting mission objectives must be considered. This efficiency for each experiment is simply the ratio of points assigned (Column 3) to the experiment weight (Column 2) and is tabulated as Column 4.

The instruments which are useful in both the entry and on-surface regimes are indicated by an asterisk (*). The duality of the role has not been taken into account in computing the efficiency for those instruments.

TABLE 1.3-1: ASSIGNMENT OF MISSION VALUE POINTS TO THE VARIOUS EXPERIMENTS SUGGESTED FOR THE 1969 MISSION TO MARS

Parameter or Experiment	Weight lbs.	Weight oz.	Points Assigned	Efficiency pts./lb.
Landers during entry: 10 points				
* Temperature		5	1	3.2
* Pressure		5	1/2	1.6
* Density	1	8	2	1.33
* Composition (gas chromatograph)	7	0	2	0.29
* H_20 Detector	1	8	1/2	0.33
* 0_2 Detector	1	8	1/2	0.33
* 0_3 Detector	1	8	1/4	0.16
* A Detector	1	8	1/4	0.16
* N_2 Detector	1	0	1/4	0.25
* CO_2 Detector	1	0	1/4	0.25
Radar Altimeter	15	0	2	0.13
Electron Density	3	0	1/2	0.16
TOTALS	35	2	10	0.29 (average)
Landers on Surface: 60 points				
Temperature		5	3	9.6
Pressure		5	1/2	1.6
Density		12	4	5.3
Composition (gas chromatograph)	7	0	2	0.44
H_2O Detector	1	8	1/2	0.33
0_2 Detector	1	8	1/2	0.33
0_3 Detector	1	8	1/4	0.16
A Detector	1	8	1/4	0.16
N_2 Detector	1	0	1/4	0.25

TABLE 1.3-1: ASSIGNMENT OF MISSION VALUE POINTS TO THE VARIOUS EXPERIMENTS SUGGESTED FOR THE 1969 MISSION TO MARS (cont.)

Parameter or Experiment	Weight		Points Assigned	Efficiency pts./lb.
	lbs.	oz.		
CO_2 Detector	1	0	1/4	0.25
Wind speed and direction	2	0	2	1.0
Television panorama	10	0	10	1.0
Precipitation	1	0	1/2	0.50
Surface sounds		8	3	6.0
Light levels		5	1/2	1.6
Surface penetrability	1	0	2	2.0
Soil moisture	2	0	1	0.5
Seismic activity	8	0	1/2	0.06
Surface gravity	3	0	1/2	0.16
Radioisotope growth detector	6	0	3	0.5
Turbidity and pH growth detector	4	0	3	0.75
Multiple chamber growth detector	4	0	10	2.5
Photoautotroph detector	3	0	3	1.0
Microscopic analysis	25	0	6	0.24
Drill, pulverizer, etc.	50	0	3 1/2	0.07
TOTALS	136	3	60	0.44 (average)

TABLE 1.3-1: ASSIGNMENT OF MISSION VALUE POINTS TO THE VARIOUS EXPERIMENTS SUGGESTED FOR THE 1969 MISSION TO MARS (cont.)

Parameter or Experiment	Weight lbs.	Weight oz.	Points Assigned	Efficiency pts./lb.
Orbiter (1000 x 19000 nautical mile orbit): 30 points				
Infrared radiation flux	3		1	0.33
Infrared radiation spectrum	29		1	0.03
Magnetic field	5		2	0.40
Television (multicolor, stereo)	125	8	20	0.16
Micrometeoroids	8		1	0.12
Bistatic radar	13		1 1/2	0.12
Charged particle flux	5	8	1	0.16
Planetary albedo	3		1	0.33
Polarimeter	6		1/2	0.08
Ultraviolet radiometer	6		1	0.16
TOTALS	204	0	30	0.15 (average)

* Instruments useful in both entry and on-surface regimes

Although the rate at which data are obtained has a relatively small part in the definition of mission success, there are important reliability considerations in the time required for the measurements, as discussed. Since instrument and systems failures are cumulative, it is very desirable to obtain the data as rapidly as possible once the instruments are properly deployed.

The rate of data acquisition anticipated for the Mars 1969 mission can be appreciated from the curve of Figure 1.3-1, in which mission value is plotted as a function of time after start of measurements, the time scale being logarithmic. The mission value scale is based on 100 percent mission value for a single Lander, thereby making the scale for the present two-Lander system appear somewhat artificial. A total possible mission value of 165 percent at the end of six months is obtained by the following apportionment:

Orbiter - three months in orbit	30%
First Lander during entry	10
First Lander on surface	60
Second Lander during entry	5
Second Lander on surface	60
TOTAL	165%

Since this quest for life is the first objective of Voyager, the exploration of Mars should be emphasized in the first missions. If the existence of life on Mars is verified, there will still be a strong impetus to determine the characteristics of that life. In that case, Mars missions should still be accented, although the problems become of more scientific and less emotional importance and should therefore take their place among the other scientific problems. If the existence of life on Mars is not verified on the first missions, there will be no particular reason to abandon the search, although it will be greatly downgraded in priority.

The need for information oriented toward the manned exploration of Mars provides an important secondary objective for the Voyager program and will still provide a higher priority for Mars missions than for those to Venus.

The above remarks have been made with the tacit assumptions that the surface temperature of Venus is indeed as high as present data indicate and that the probability of life on Venus, except possibly in the free atmosphere itself, is very low. If the assumptions are refuted by further measurements, the priorities for Voyager missions should be reviewed in the light of the new information. A comparison of the scientific interest in the two planets is difficult, as they are both packed with very interesting scientific problems and even puzzles. For instance, the nature and cause of the blue haze of Mars, the characteristics and origin of the "canals" of Mars, the energy transfers in and through the atmosphere of Venus, and the genesis and properties of the cloud layer or layers of Venus are a few of the problems demanding an explanation. Realistically, one can say only that, aside from biological aspects, Voyager missions should have equivalent scientific value for the two planets.

1.4 RECOMMENDED MISSION CONFIGURATION

1.4.1 VEHICLE COMBINATIONS

In selecting the most logical configurations for the missions possible in the Voyager program, one must make a great many judgements. It is obvious that the straightforward approach to life detection on Mars requires one or more vehicles which land on the surface, whereupon the question arises, how many? From an engineering standpoint, a large weight possibility such as that postulated for Voyager provides a wide range of Orbiter-Lander combinations without a very great change of mission efficiency. The decision as to the most feasible configuration in that case is based largely on relative scientific merit of the resulting data and on the inherent reliability obtainable.

Since reliability aspects are covered elsewhere in this report, it is sufficient for present purposes to point out that reliability has been a serious consideration throughout this study and has played an important role in final selection of the configurations for the different missions.

The scientific value of the obtainable data is more difficult to assess. Which, for instance, is the more valuable - an infrared radiation measurement from an Orbiter or a penetrability determination on the surface? Which is more important - a life detection experiment performed at two different locations on the planet or two life detection experiments performed at one location? Even though the basic objectives of the missions are well defined, there are intangibles involved in determining answers to these questions.

In making the judgments one must consider the fact that we know very little about conditions in which the measurements are to be made. Terrain features of all scales are unknown; data on atmospheric densities and circulations are marginal; information on the distribution of biologically favorable environments is needed. Because of these uncertainties, a very conservative approach has been taken in the present study. A dual Lander configuration has been taken for the Mars missions, in order to provide a two-location search for life and to minimize the possibility of complete Lander failure due to entry and surface orientation problems. The overall reliability is likewise enhanced by complete redundancy by the use of the dual-Lander system.

The specific mission configurations which have resulted from the present study are shown in Table 1.4-1, in which total weights and weights of the instrument payloads are tabulated for the vehicles recommended for the various opportunities. As indicated above, the standard configuration for Mars makes use of dual Landers, plus an Orbiter whenever feasible. The Venus 1970 and 1972 format represents a variance from this standard in that only a single Lander is indicated. This comes about on the Venus 1970 mission because the prime objective for that opportunity is to obtain a high resolution radar map of the planet from the Orbiter. The single Lander is in a sense a bonus, since it can be carried along within the weight allowance and its successful operation is not required in obtaining the radar map. This should not be construed as minimizing the value of the measurements from the Lander, however. Some valuable measurements, as indicated by the experiment lists which follow, can be made from that one Lander.

The single Lander configuration for Venus 1972 is dictated by the necessity for thermal protection. A 6.5 hour operating life on the surface is about the minimum acceptable for that mission, because of communications requirements. Protection of the operating components from the high surface temperature is much more efficiently accomplished for one Lander than for two Landers. This, of course, degrades both the reliability and scientific return from that for the dual-Lander configuration, but it appears to be necessary.

TABLE 1.4-1: MISSION FORMATS WHICH HAVE BEEN DERIVED AS A RESULT OF THE PRESENT VOYAGER STUDY.

OPPORTUNITY	ORBITER		LANDER #1		LANDER #2	
	Total Weight (lbs.)	Payload (lbs.)	Total Weight (lbs.)	Payload (lbs.)	Total Weight (lbs.)	Payload (lbs.)
Mars 1969	2058	215	1450	155	1450	155
Mars 1971	2100	223	2000	255	2000	255
Mars 1973	1400	77	2000	255	2000	255
Mars 1975	--	--	2000	255	2000	255
Venus 1970	2145	137	525	60	--	--
Venus 1972	1800	61	2600	210	--	--

Another important consideration is the apportionment of weight allowance to the Orbiter versus that for the Landers. From the economics, it is inefficient to put weight into orbit, because of the propulsion energy required to do so. On the other hand, the advantages of using an Orbiter as a communications relay are substantial, and an orbiter provides measurement possibilities which are denied otherwise.

The use of an Orbiter is strongly emphasized in the 1970 Venus mission because of the desirability of obtaining a high resolution radar map of the planet. The radar mapping equipment is massive and has high power requirements but the cloud cover prohibits this information being obtained by optical means. The format proposed for the first two Mars missions reflects the strong emphasis on life detection and surface measurements. Even the payload of the Orbiter is oriented toward surface characterization, most of the weight being taken up by the camera system for color and stereoscopic picture of the surface. In the cases such as those of Mars 1973 and Venus 1972 in which surface properties as seen from the Orbiter receive less emphasis, the Orbiter payload constitutes only a small portion of the total. Investigations of the upper atmosphere of Mars which are of primary importance for manned entry, are delayed until the 1973-1975 period. They are performed from both Landers and Orbiters, but the greater efficiency for Lander weight is utilized by emphasizing instrumented Landers for the necessary measurements.

1.4.2 ANALYSES OF SPECIFIC MISSIONS

A. Mars 1969

The Mars 1969 mission has been defined more fully in this study than have the others, and it can be thought of as indicating the general philosophy for the whole program. A vehicle of 2058 pounds will be injected into an orbit around the planet, and two identical Landers of 1450 pounds each will enter the atmosphere and make a soft landing on the surface. Both the Orbiter and the Landers will be instrumented for obtaining scientific measurements. Since primary communication with the Earth will be through the Orbiter, data obtained from the Landers will be relayed back to Earth by way of the Orbiter. A low data-rate direct transmission system between Earth and Landers can be activated in case of failure of the Orbiter system.

The measurements divide themselves conveniently into three domains; namely, measurements from the Landers during their entry into the atmosphere, measurements from the

Landers while they are resting on the surface, and measurements taken from the Orbiter after it is injected into orbit around the planet. A fourth set of measurements, in-transit measurements of the interplanetary environment, are anticipated, but they are relatively straightforward and have received little emphasis in the study.

The weights of the instruments for the three types of measurements are approximately the following:

Each Lander during entry	35 pounds
Each Lander on surface	126 pounds
Orbiter	215 pounds

The total instrument payload weight is approximately 525 pounds. Some of the instruments are used both during entry and on the surface, a fact which accounts for the apparent discrepancy between total instrument weight and that indicated in the breakdown. It should be pointed out that these weights are for the instruments themselves, and do not include power supplies, telemetry components, and other items involved in the engineering functions of the vehicles. Some sample-handling equipment for use on the Landers is included, however.

The measurements for each Lander during the atmospheric entry phase are listed in Table 1.4.2-1. Included in the tabulation are the sizes, weights and power requirements of the instruments by which the measurements are to be made. Emphasis is on the basic types of measurements necessary to obtain a good vertical profile of the atmosphere; namely, temperature, pressure, density, and composition, along with altitude at which the data are taken. By measuring all of the state parameters, one has an over-determined system, which permits valuable checks for internal consistency of the data. The only other measurement for the entry phase is electron density.

TABLE 1.4.2-1: EXPERIMENTS FOR THE ENTRY AND DESCENT OF THE 1969 LANDERS INTO THE ATMOSPHERE OF MARS.

Parameter or Experiment	Size	Wt lbs	Power Watts	Inst. * No.
1. Temperature	1 in. dia. x 3 in. long	0.3	0.07	(I-24)
2. Pressure	1.6 in. dia. x 2 in. long	0.3	0.10	(I-17)
3. Density		1.5	2	(I-20)
4. Composition				
a. Mass spectrometer	10 in. x 5 in. x 3 in.	6	6	(I-43)
b. Gas Chromatograph	5 in. x 5 in. x 8 in.	7	4.5	(I-8)
5. Altitude				
Radar Altimeter	5 in. x 10 in. x 10 in. + 12 in. antenna 6 in. x 6 in. x 3 in. electronics	15	25	(I-5)
6. Electron Density (Langmuir Probe)	1 in. dia. x 6 in. long sensor in free stream	3	3	(I-39)

* The instrument number refers to the instruments listed and described in Section 1.9 of this volume.

There are some serious engineering problems in obtaining most of these data at high altitudes in the atmosphere. Only electron density measurements are feasible before the vehicle reaches sub-sonic speeds, so measurements of the state parameters will be restricted to the lower portions of the atmosphere. Furthermore, present uncertainties of the density distribution and total mass of the atmosphere itself makes it very difficult to estimate the altitude at which the vehicles will reach sub-sonic speed. Hopefully, the Mariner program will supply the needed data before this first Voyager mission. Pending these data, a conservative attitude has been taken in the present study, the philosophy being that the probability of the vehicles surviving entry will not be jeopardized by the effort to obtain high altitude data.

Some inferences about the atmospheric profile can be made from accelerometers and other diagnostic instrumentation aboard the vehicles which have not been considered as part of the scientific instrument payload. Atmospheric density and density gradients are subjects for this type of interpretation.

A great deal of emphasis throughout the study has been on surface measurements, the primary objective being the detection and characterization of life on Mars. This emphasis shows up in the instrument payloads for the Landers, a definition of which constitutes Table 1.4.2-2.

TABLE 1.4.2-2: EXPERIMENTS FOR EACH OF THE DUAL LANDERS OF THE MARS 1969 MISSION, THE MEASUREMENTS TO BE MADE AFTER THE VEHICLES COME TO REST ON THE SURFACE.

Parameter or Experiment	Size	Wt lbs	Power Watts	Inst. No.
1. Temperature	1 in. dia. x 3 in. long	0.3	0.07	(I-24)
2. Pressure	1.6 in. dia. x 2 in. long	0.3	0.10	(I-17)
3. Density		1.5	2	(I-20)
4. Composition of Atmosphere				
a. Mass spectrometer	10 in. x 5 in. x 3 in.	6	6	(I-43)
b. Gas Chromatograph	5 in. x 5 in. x 8 in.	7	4.5	(I-8)
5. Wind speed & direction	5 in. dia. x 6 in. long	2	0.5	(I-67)
6. Television - 2 cameras	(1 panorama & 1 microscope)	20 (10 ea.)	40 (20 ea.)	
7. Precipitation	3 in. dia. x 3 in. long	1	1	(I-36)
8. Surface Sounds	2 in. dia. x 3 in. long includes electronics	0.5	1	(I-34)
9. Light level indicator	2 in. x 2 in. x 2 in.	0.3	0.1	(I-84)
10. Surface penetrability	1 in. x 2 in. x 6 in.	1	0.1	(I-25)
11. Soil moisture	3 in. x 3 in. x 10 in.	2	25	(I-70)
12. Seismic activity (1-axis seismometer)	5 in. dia. x 6 in. long	8	1	(I-21)
13. Surface gravity	5 in. x 5 in. x 3 in.	3	3	(I-72)
14. Radioisotope growth detector	8 in. x 10 in. x 6 in. + four 1 in. x 2 in. x 3 in.	6	3	(I-19)
15. Turbidity & pH growth detector	6 in. x 6 in. x 6 in.	4	1	(I-53)
16. Multiple chamber growth detector	approx. 6 in. x 6 in. x 8 in.	4	2	(I-54)
17. Photoautotroph detector	2 in. x 2 in. x 10 in.	3	1	(I-62)
18. Microscopic Analysis	12 in. x 5 in. x 5 in.	15 (excluding TV #2)	7	(I-71)
a. atmospheric aerosols				
b. surface materials				
c. biological materials				
19. Drill		20		
20. Pulverizer		10		
21. Sample handling equipment		20		

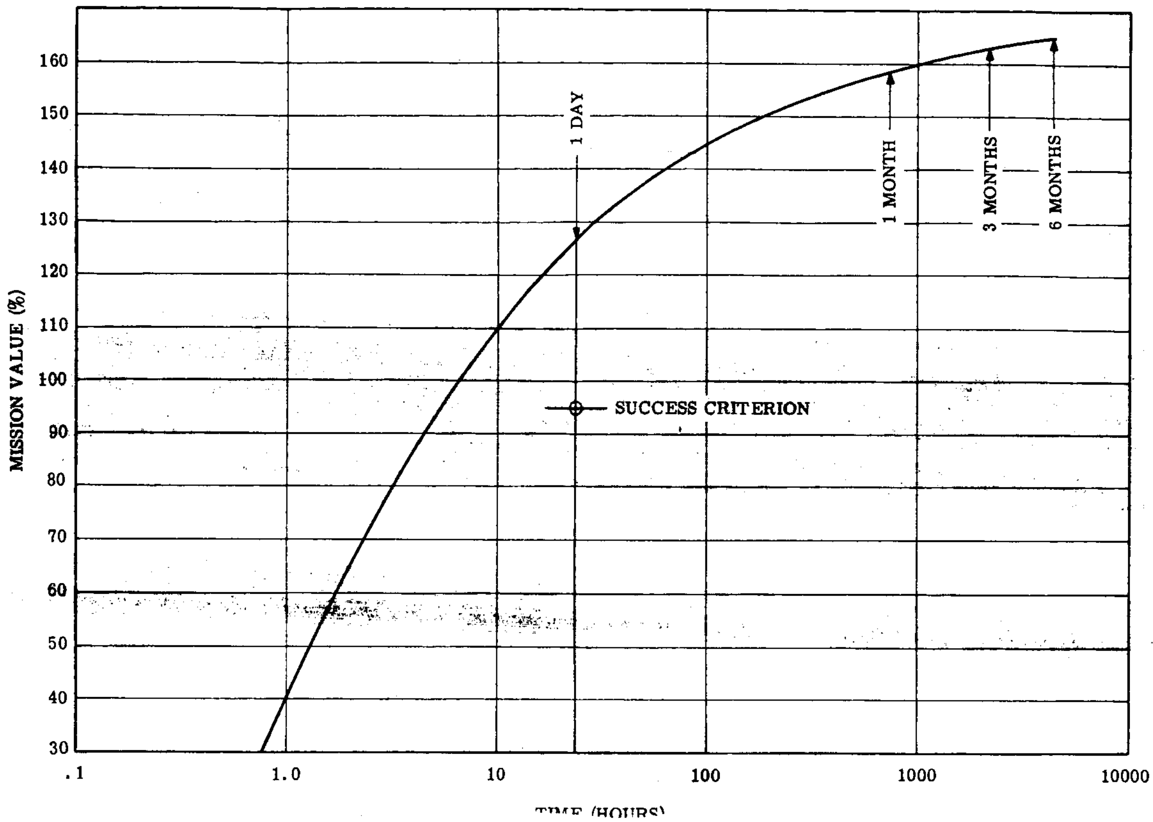


Figure 1.3-1 Rate at Which Scientific Value at the 1969 Mars Mission is Attained After Deployment of the Instrumentation. The Total of 165% Mission Value Arises Because of Duplication of Landers.

It is evident from the curve that according to the time evaluation of data acquisition from the various instruments which was assigned here, a major portion of the mission can be accomplished by a one-day successful operation of all of the instruments and systems. In fact, the arbitrary criterion of mission success will have been satisfied by a 75 percent successful operation for the first day. As shown in Section 4.0 of this Volume, it is unlikely that even a 75 percent successful one-day operation will be attained, but the success criterion can still be satisfied by a longer period of operation at less than 75 percent perfection.

Figure 1.3-1 also shows that there is relatively little scientific value to be gained in the last half of the six-months operating period. This should not be taken to indicate that the data obtained for longer periods are not important. They obviously are important in many respects, and particularly in studies of changes of the parameters with time. However, for satisfying the specific objectives set up for this particular mission, the latter half of the period does not contribute very much.

It is anticipated that the exploration of both Mars and Venus will be accomplished during the course of the Voyager program, so one is faced with assigning relative priorities to the explorations. Limitations of launching facilities, as well as on funds and manpower, makes such assignments necessary.

In making a judgement as to the relative emphasis for Mars versus Venus, the possibility of finding life is undoubtedly the most important criterion. From a scientific point of view the existence of life on another planet, with all that can be inferred therefrom, is obviously of first rate significance. Perhaps more important, however, is the impact that the discovery would have on the society as a whole, probably more from an aesthetic than from a scientific standpoint.

Exo-biologists generally agree that a great deal about the existence and character of life can be inferred from a knowledge of the environment in which the life develops. Thus, every effort is made to obtain consistent and repeated measurements of the temperature, pressure, density, and composition of the atmosphere. Soil mositure content and the composition and penetrability of surface material also fall into the category of biological environmental data. All of these measurements are valuable in contexts other than biological, of course.

Experiments oriented more directly toward life detection are headed by the television system, which is designed to give a full 360° panorama of the area surrounding the Lander location, and by the microphone for picking up sounds generated in the vicinity. These types of information have a greater emotional appeal than perhaps any other to be obtained in the whole program, and it may well be that the first picture taken from the surface of Mars will satisfy all of the life-detection requirements. In that case, biologists will be able to concentrate all of their efforts in the characterization of that life, as opposed to its detection.

Microscopic examination of samples of the surface material and atmospheric aerosols are in somewhat the same category as the television panorama,in that visual pictures will be obtained. However, one would probably have to be very fortunate to find in a sample of surface material an object which can be definitely identified as a living organism. His chances would be greatly enhanced by appropriate concentration techniques, but this type of sample processing is anticipated only for later missions. The microscope can, however, be used to detect turbidity changes or other manifestations of biological activity.

The purely biological experiments for the Mars 1969 mission include the most attractive of those which have been developed and which are simple enough for reliable remote operation. The term "growth detector" used in the tabulation may be misleading in that a positive signal from the experiment may not be dependent on only growth. Dr. Lederberg has pointed out that metabolic activity may be more easily detected than is growth. A turbidity change would be most easily accomplished by a change of numbers of organisms, but carbon dioxide effluence might well be the result of only metabolic activity.

A great deal of reliance has been put on the multiple chamber experiment developed by Dr. Lederberg and his associates. Versatility is extremely important in the unknown environmental conditions and versatility is the key word in the multiple chamber experiment. Considerable further work is needed to develop assays for use in the experiment, but the basic concept is such as to be able to take advantage of the assays once they are developed. According to Dr. Lederberg, a number of good experiments will be available in plenty of time for use on Voyager.

The instrument listed as a photoautotroph detector also requires development, although the principle on which it is based seems straightforward enough.

A serious problem still exists in the sterilization of many of the biological experiments. High temperatures degrade the materials of which the assays consist. The problem is only mentioned here since sterilization aspects for the whole project are discussed in Volume V.

Planetary characterization experiments for this first Voyager mission include a gravity measurement and a single-axis seismometer, in addition to the microscope, penetrometer, soil moisture measurement, and panoramic television mentioned above. The mineralogical measurements such as X-ray diffraction, α particle scatter, and neutron activation are introduced on later missions.

Atmospheric measurements for the Landers consist, in addition to the basics of temperature, pressure, density, and composition, of surface wind direction and speed, any possible precipitation, either liquid or solid, and a microscopic analysis of atmospheric aerosols. An upper atmospheric measurement of the electron density profile is included for the entry phase of the Landers.

Experiments for the Orbiter for the 1969 Mars mission are listed in Table 1.4.2-3. The emphasis is on obtaining as much information on the various geologic provinces of the planet as possible. A complete stereoscopic map of a large portion of the southern hemisphere can be obtained by a pair of vidicon cameras. The resolution of the map will vary between 1000 meters at the periapsis point to about 3000 meters at the point 90° from the periapsis, and will permit a topographic map showing mountain ranges, large depressions, and escarpments. Pictures obtained from three image orthicon cameras will give an order of magnitude higher resolution and can be superimposed to provide color, but the aerial coverage is correspondingly reduced. Finally, one image orthicon of only 20 meter resolution will give great detail in very small areas of the planetary surface.

TABLE 1.4.2-3: EXPERIMENTS FOR THE ORBITER OF THE MARS 1969 MISSION

Parameter or Experiment	Size	Weight (lbs)	Power (Watts)	Inst. No.
1. IR Multichannel Radiometer	5 in. x 7 in. x 4 in.	3	3	(I-2)
2. IR Spectrometer	12 in. dia. x 15 in. long	29	7	(I-1)
3. Magnetic field Magnetometer	4 in. x 4 in. x 6 in	5	5	(I-23)
4. Television (multicolor, stereo)		125.5	25	
5. Cosmic Dust	5 in. x 5 in. x 5 in.	2.5	.2	(I-37)
6. Ionospheric profile Bi-static radar	4 in. x 4 in. x 12 in. + 10 ft. & 3 ft. dipoles	13	2	(I-85)
7. Ionization chamber and GM tube assembly	6 in. dia. x 12 in. long	5.5	1	(I-12)
8. Solar Multichannel Radiometer	5 in. x 7 in. x 4 in.	3	3	(I-79)
9. Polarimeter (skylight analyzer)	7 in. dia. x 5 in. long	4.5	4.5	(I-68)
10. Sferics	4 in. x 5 in. x 6 in. plus whip antenna	3	2	(I-82)
11. X-ray flux from sun	3 in. x 3 in. x 6 in.	3	1	

A number of radiometric experiments are included in the Orbiter payload. Multi-channel radiometric measurements of the planet-reflected and planet-emitted rays will permit analyses of important energy budgets and atmospheric processes, and a polarimeter operating in the solar wavelength range will provide data on scattering and reflection properties of the atmosphere and surface materials. Selected narrow bands in the far ultraviolet will give photochemical information about the upper atmosphere. Bi-static radar signals received at the Orbiter will permit valuable inferences regarding the ionospheric density profile and magnetic fields through which the signals are transmitted, and will provide data on radar reflectivity of the surface material. Additional surface characteristics, such as electrical conductivity and surface density profiles, can perhaps be inferred from the bi-static radar data.

Direct measurements of the magnetic fields at orbit altitude will be obtained by the three-axis magnetometer, and charged particle flux measurements will be made by means of a combination Geiger tube-ion chamber assembly. More sophisticated particle measurements, including both energy spectra and particle direction, are not included in this first mission, the thought being to keep the equipment simple for determining the existence and orders of magnitude of trapped particles. Spectrometers and telescopes based on these preliminary data are included for later missions.

Finally, a micrometeoroid detector completes the Orbiter instrument payload.

The Voyager flights, being of long duration and covering great distances, provide a good opportunity for interplanetary environmental measurements. However, the Mariner program has already provided a few data on the characteristics of magnetic fields and particle fluxes between the Earth and Venus, and other Mariner flights will go to Mars before the first Voyager mission is launched. Deep space probes have already been projected into far distances and various types of orbiting observatories are now or will shortly be furnishing large numbers of data on the interplanetary characteristics.

For these reasons, the in-transit measurements for Voyager have been given only a secondary role. The only in-transit measurements included for the Mars 1969 mission are micrometeoroids, magnetic fields, and charged particle fluxes. These data will augment our knowledge about the spatial and temporal variations of solar emissions and their interaction with the nagnetic fields of interplanetary space, without detracting appreciably from the primary objective of planetary exploration.

One or two experiments of particular note which could be, but so far have not been, incorporated into the Voyager program are those of cosmological interest. Dr. Ralph Alpher has emphasized the value of a gamma ray flux and spectral measurement in studies of the distribution of matter in and between galaxies, a quantity which is of considerable interest in cosmological theories. A second interesting possibility is a measurement of time dilatation and the "red shift" predicted in relativity considerations. According to Dr. Alpher, a frequency standard of the required high stability is commercially available. An outline of Dr. Alpher's ideas is included in an appendix to this report.

Additional data of scientific interest which can be obtained from the Voyager flight, but for which no particular "experiment" is required, are those obtained from an analysis of the flight parameters of the mission and those deduced from an analysis of the electromagnetic signals received from the spacecraft. Planetary mass and mass distribution fall in the first of these bonus data, while interplanetary magnetic fields and electron densities constitute the second category. The details of these analyses are included in the attached appendices.

Emphasis for the 1969 Mars Orbiter is on obtaining television pictures of the planetary surface. The details of the seven cameras which have been integrated into the Orbiter payload are listed as Table 1.4.2-4. The resolutions given apply at the periapsis point and therefore represent the maximum resolution.

The two vidicon cameras oriented at about 20° on opposite sides of the nadir direction will be synchronized to give a steroscopic map of the surface. A height resolution of approximately 345 meters is obtainable from the stereoscopic pair and about 40 percent of the planetary surface will be mapped in 30 days operation.

TABLE 1.4.2-4: SPECIFICATIONS OF TELEVISION EQUIPMENT IN THE MARS 1969 ORBITER PAYLOAD

Type Camera	Resolution at Surface	Direction of View	Color
Vidicon	1000 meters	15° forward of nadir	B & W
Vidicon	1000 meters	15° to rear of nadir	B & W
Vidicon	1000 meters	toward nadir	B & W
Image Orthicon	140 meters	toward nadir	red
Image Orthicon	140 meters	toward nadir	yellow-green
Image Orthicon	140 meters	toward nadir	blue
Image Orthicon	20 meters	toward nadir	B & W

The third vidicon camera is oriented toward the nadir and will be used only in the portion of the Orbit in which the field of view of one of the stereoscopic cameras is above the planet's horizon. This occurs in the region near the terminator, in which case shadows will give an indication of topography.

Higher resolution pictures are obtained by image orthicon cameras, the fields of view of which are all nested around the nadir direction. Superposition of the pictures from the optically-filtered image orthicons will give color pictures, thereby permitting better delineation of geological provinces and perhaps biological domains as well. High resolution black and white pictures obtained from the final image orthicon camera will permit analysis of the small scale features of very restricted areas of the surface. It is obvious, of course, that the image orthicon pictures do not constitute complete coverage of the surface area.

Infrared radiation instruments on the Orbiter include a spectrometer, the main purpose of which is to confirm the existence of the Sinton absorption bands of organic molecules, and a multi-channel radiometer for measurement of planetary emissions in selected spectral regions such as the 6.7 μ band of water vapor, the 15 μ band of carbon dioxide, and the 8-12 μ water vapor window. A similar multi-channel radiometer operating in the near UV-visible-near IR region will provide data for inferring atmospheric and surface characteristics from their light-scattering and reflecting properties, and will permit a determination of the planetary albedo. The polarimeter data will supplement the data from this latter radiometer and make the inferences more specific.

The data from the bi-static radar can be combined with that from the magnetometer to infer the electron densities and magnetic fields which exist between the Orbiter and planetary surface, and the radar data themselves provide information about the surface properties.

On this first Voyager mission, it does not seem logical to try to measure energy spectra or direction of Van Allen-type radiation, since the mere existence of Van Allen belts for Mars is not yet established. The simple and reliable flux measurements from the Geiger tube-ion chamber assembly can be combined with the nagnetic field measurements to obtain the general character of such belts, if they indeed exist. These data can then be used to set energy ranges for the more sophisticated instruments of later missions.

The micrometeoroid detector, the data from which will be of great interest, completes the Orbiter instrument payload.

B. Mars 1971, 1973 and 1975 Missions

In the interests of brevity, the suggested payloads for later missions to Mars will not be analyzed here in as great detail as that for the 1969 mission. A few comments however are included.

It has been the objective throughout the study to incorporate evolutionary development into the scientific program. The evolution may be thought of in any one of three senses. First, the fund of knowledge on which the instruments are based will increase with time, thereby permitting measurements to be more specific. Mariner data will be available even before the 1969 period , and Earth-based observations are rapidly improving as further work is done. This increased knowledge will no doubt modify payload specifics even before the 1969 mission is launched. The knowledge will be tremendously advanced by the first successful Voyager mission, and later missions will be modified accordingly. A recognition of this information advance has conditioned plans for later payloads. For instance, the inclusion of charged particle spectrometers and telescopes on the 1971 and 1973 Orbiter payloads is contingent on the general character of the radiation belts being determined before that time.

The same anticipated advance of information, mainly from the Voyager program itself, is the basis for a shift of emphasis in later missions. Life detection should obviously be stressed on the first mission, and its characterization and environmental adaptation is of importance in subsequent missions. However, it seems logical that by the third mission data oriented more directly toward eventual manned missions should be started up the priority ladder. This thinking is reflected in the 1973 mission by the inclusion of meteorological rocket sondes for probing the upper atmosphere, and by a change of orbit to permit direct mass spectrometer measurements of the upper atmospheric composition. Other atmospheric measurements are also amplified for the later missions.

The second type of evolution is that represented by improved techniques of measurement. The details of this evolution are difficult to predict, more difficult even than the evolution of scientific data. The rapid rate at which techniques are improving is everywhere evident; who can predict the details of a maelstrom?

There are, however, certain techniques which show already their great potential in the Voyager project. Notable here are the many uses to which lasers are applicable. Scattering and absorption properties of a planetary atmosphere are of primary interest to meteorologists and atmospheric physicists. One can predict that the Earth's atmosphere will be laced with laser beams within a very few years. The same techniques can be applied on Mars. Similarly, the heating properties of a laser beam can be utilized for producing localized high temperature radiant emissions from materials. The spectroscopic analysis of radiation emitted from a laser-heated sample of the Martian surface will provide a great deal of information about the detailed constituents. Both of these uses of lasers are integrated into the Mars 1975 payload.

Other areas in which rapid advances are already under way are those of sample-processing and biological assays. It is believed that the instrumentation proposed for the Voyager is versatile enough that advantage can be taken of these and other new techniques as they become available.

The detailed experiments or instruments for the suggested payloads of the Mars 1971, 1973 and 1975 missions are listed as Table 1.4.2-5. The measurement domains are the entry or descent through the atmosphere, the planetary surface, and in orbit around the planet. The term entry is used for the portion of the Lander trajectory from the effective top of the planetary atmosphere to the point at which the speed of the vehicle becomes subsonic. The portion of the trajectory in which the vehicle speed is subsonic is termed the descent.

C. Venus 1970

Problems associated with the exploration of Venus are quite different from those for Mars. The high surface temperature, dense atmosphere, and complete cloud cover each has its own influence on the instrument payload packages.

Emphasis on the first Voyager mission to Venus, proposed for the 1970 opportunity, is on obtaining a radar map of the planetary surface. Of nearly equal priority is a definition of the atmospheric properties, measurements for which include the usual parameters of state during entry or descent and on the surface, extent and properties of the cloud during descent, various radiative fluxes observable from orbit, and the ionospheric properties as possible by the use of bi-static radar. Until further definition of the temperature distribution on the planet is obtained, it does not seem feasible to conduct a search for life there.

A list of the suggested experiments for the 1970 mission is given as Table 1.4.2-5. It should be pointed out that the single entry capsule anticipated for this mission is of very limited payload capacity, its total weight being only 525 pounds, and it is designed for only about 10 minutes useful life on the high-temperature surface.

D. Venus 1972

The final payloads analyzed in detail in the present study are those for the Orbiter-Lander configuration for the Venus 1972 opportunity. The suggested instruments or experiments are listed as Table 1.4.2-6. The major emphasis for this case is on determining surface characteristics, although the atmosphere is not neglected. The Landers are designed for 6.5 hours operating life on the surface (for the worst model atmosphere), thereby providing enough time for sample analysis of surface materials and atmospheric aerosols and for observing significant changes of wind direction and speed.

TABLE 1.4.2-5: EXPERIMENTS FOR VOYAGER MISSIONS TO MARS FOR THE OPPORTUNITIES OF 1971, 1973 AND 1975.

MARS 1971

Parameter or Experiment	Size	Wt lbs	Power Watts	Inst. No.
I. Landers During Entry or Descent				
1. Temperature	1 in. dia x 3 in. long	0.3	0.07	(I-24)
2. Pressure	1.6 in. dia. x 2 in. long	0.3	0.10	(I-17)
3. Density		1.5	2	(I-20)
4. Composition				
a. Mass Spectrometer	10 in. x 5 in. x 3 in.	6	6	(I-43)
5. Altitude Radar Altimeter	5 in. x 10 in. x 10 in. plus 12 in. dish antenna	15	25	(I-5)
6. UV Multichannel Radiometer	3 in. dia. x 5 in. long	1.5	1.5	(I-78)
1215 Å (Lyman α), 1026 Å (Lyman β), 972 Å (Lyman γ), 584 Å (HeI), 304 Å (HeII)	sharp filters			
1445 Å - 1500 Å, 2500 Å - 3000 Å	band filters			
7. 8446 Å Radiometer	3 in. dia. x 5 in. long	0.3	0.2	(I-40)
II. Landers on Surface				
1. Temperature	1 in. dia. x 3 in. long	0.3	0.07	(I-24)
2. Pressure	1.6 in. dia. x 2 in. long	0.3	0.10	(I-17)
3. Density		1.5	2	(I-20)
4. Composition				
a. Mass Spectrometer	10 in. x 5 in. x 3 in.	6	6	(I-43)
b. Gas Chromatograph	5 in. x 5 in. x 8 in.	7	4.5	(I-8)
5. Wind Speed & Direction	5 in. dia. x 6 in. long	2	0.5	(I-67)
6. Television - 2 cameras (1 panoramic & 1 microscope)		20 (10 ea.)	40 (20 ea.)	
7. Surface Sounds	2 in. dia. x 3 in. long includes electronics	1		(I-34)
8. Polarimeter (skylight analyzer)	6 in. dia. x 3 in. long	4.5	4.5	(I-68)
9. Surface Penetrability	1 in. x 2 in. x 6 in.	1	0.1	(I-25)

TABLE 1.4.2-5 (Cont'd)

Parameter or Experiment	Size	Wt lbs	Power Watts	Inst. No.
10. Seismic Activity 1-Axis Seismometer	5 in. dia. x 6 in. long	8	1	(I-21)
11. X-ray Diffractometer	10 in. x 10 in. x 10 in.	10	15	(I-32)
12. α- particle scattering				
Sensor	3 in. x 3 in. x 3 in.	7	2	(I-57)
Electronics	5 in. x 7 in. x 7 in. plus storage			
13. Thermal Diffusivity of ground	3 in. x 3 in. x 3 in. plus scraper	1	25	(I-64)
14. Electrical conductivity of ground	3 in. x 3 in. x 3 in.	1	1	(I-65)
15. Microscopic Analysis a. atmospheric aerosols b. surface material c. biological material	12 in. x 12 in. x 5 in.	15 (excluding TV #2)	7	(I-71)
16. Multiple chamber growth detector	6 in. x 6 in. x 8 in.	4	2	(I-54)
17. Photoautotroph detector	2 in. x 2 in. x 10 in.	3	1	(I-62)
18. (Insolation) Pyrheliometer	3 in. x 3 in. x 3 in.	1	1	(I-16)
19. Surface radioactivity	200 in.^{3}	8	2	(I-13)
20. Meteor trails	6 in. x 6 in. x 6 in.	2.5	2.5	(I-5)
21. Ionospheric Profile: Bottomside sounder	12 in. x 12 in. x 12 in. + antenna	50	25	(I-87)
22. Sferics	4 in. x 5 in. x 6 in. plus whip antenna	3	2	(I-82)
23. Eclipse by Phobos	Use TV + special filter	1	0.3	
24. Insect attractor	1 in. x 1 in. x 1 in.	.1	1	(I-69)
25. Pulse Light	5 in. dia. x 2 in. long	1	0.1	(I-75)
26. Drill		20		
27. Pulverizer		10		
28. Sample handling equipment		20		
III. Orbiter				
1. IR Multichannel radiometer	5 in. x 7 in. x 4 in.	3	3	(I-2)
2. Solar multichannel radiometer	5 in. x 7 in. x 4 in.	3	3	(I-79)
3. Magnetic field	4 in. x 4 in. x 6 in.	5	5	(I-23)
4. Electron spectra and direction	4 in. x 4 in. x 8 in.	2	1	(I-10)
5. Proton spectra and direction	4 in. x 4 in. x 12 in.	3	1	(I-11)
6. Television (multi-color; stereo)	2000 in.^{3}	125.5	25	
7. Cosmic Dust	5 in. x 5 in. x 5 in.	2.5	.2	(I-37)
8. Ionospheric profile: Bi-static radar	4 in. x 4 in. x 12 in.	13	2	(I-4)
9. Altitude (Radar Altimeter)	5 in. x 10 in. x 10 in. + 12 in. dish	15	25	(I-5)
10. UV multichannel radiometer	3 in. dia. x 5 in. long	1.5	1.5	(I-78)

TABLE 1.4.2-5. (Cont'd)

Parameter or Experiment	Size	Wt lbs	Power Watts	Inst. No.
11. UV Solar Spectrometer	9 in. x 10 in. x 20 in.optics +6 in. x 10 in. x 6 in. elec.	22	12	(I-81)
12. Sferics	4 in. x 5 in. x 6 in. plus whip antenna	3	2	(I-82)
13. Faraday cup	36 in. x 10 in. x 10 in.	20	4	
14. X-ray flux from sun	3 in. x 3 in. x 3 in.	3	1	

TABLE 1.4.2-5 (Cont'd)

MARS 1973

I. Landers During Entry or Descent					
	Parameter or Experiment	Size	Wt lbs	Power Watts	Inst. No.
	1. Temperature	1 in. dia. x 3 in. long	.3	.07	(I-24)
	2. Pressure	1.6 in. dia. x 2 in. long	.3	.1	(I-17)
	3. Density		1.5	2	(I-20)
	4. Gas Chromatograph	5 in. x 5 in. x 8 in.	7	4.5	(I-8)
	5. Altitude (radar altimeter)	5 in. x 10 in. x 10 in. + 12 in. dish	15	25	(I-5)
	6. UV Solar spectrum	9 in. x 10 in. x 20 in.optics 6 in. x 10 in. x 6 in. elec.	22	12	(I-81)
	7. Electron density (Langmuir Probe)	6 in. x 6 in. x 3 in. elec. 1 in. dia. x 6 in. long probe in freestream	3	3	(I-39)
II. Landers on Surface					
	1. Temperature	1 in. dia. x 3 in. long	.3	.07	(I-24)
	2. Pressure	1.6 in. dia. x 2 in. long	.3	.10	(I-17)
	3. Density		1.5	2	(I-20)
	4. Gas Chromatograph	5 in. x 5 in. x 8 in.	7	4.5	(I-8)
A*	5. Wind Speed & Direction	5 in. dia. x 6 in. long	2	0.5	(I-67)
A=2, B-1	6. Television		(10 ea.)	(20 ea.)	
A	7. Surface sounds	2 in. dia. x 3 in. long includes preamp.	1	1	(I-34)
A	8. Microscopic analysis				
	a. atmospheric aerosols				
	b. surface material	12 in. x 12 in. x 5 in.	15	7	I-71)
	c. biological material				
	9. Multiple chamber growth detector	6 in. x 6 in. x 8 in.	4	2	(I-54)
	10. Photoautotroph detector	2 in. x 2 in. x 10 in.	3	1	(I-62)
A	11. Meteor ionization trails	6 in. x 6 in. x 6 in. Same as I-5 with added electronics	2.5	2.5	(I-5)
A	12. Ionospheric profile Bottomside sounder	12 in. x 12 in. x 12 in. + antenna	50 + ant.	25	(I-87)
B	13. Rocket soundings of atmosphere(8 rockets)	5 lbs. payload 20 lbs. total wt. per rocket. 4 in. dia. x 20 in. } Includes receiver on Lander	190	25	
A	14. Seismic properties				
	a. natural	10 in. dia. x 15 in. long	34	4	(I-91)
	b. induced	300 in.^{3} + geophones & explosives	90 (Inc. explosives & geophones)	5	(I-90)
	15. Polarimeter - Skylight Analyzer	6 in. dia. x 3 in. long	1.5	1.5	(I-68)
	16. Insect Attractor	1 in. x 1 in. x 1 in.	.1	1	(I-69)
	17. Pulse Light	5 in. dia. x 2 in. long	1	.1	(I-75)
A	18. Sample handling equipment		20		

*A means the item is on Lander A only
B means the item is on Lander B only

TABLE 1.4.2-5 (Cont'd)

Parameter or Experiment	Size	Wt lbs	Power Watts	Inst. No.
III. Orbiter				
1. Magnetic field magnetometer	4 in. x 4 in. x 6 in.	5	5	(I-23)
2. Proton telescope	8 in. x 8 in. x 8 in.	7	1.5	(I-11)
3. Electron telescope	8 in. x 8 in. x 8 in.	7	1.5	(I-10)
4. Mass spectrometer	10 in. x 5 in. x 3 in.	6	6	(I-43)
5. Electron probe (Langmuir Probe)	6 in. x 6 in. x 3 in. + 1 in. dia. x 6 in. long probe in freestream	3	3	(I-39)
6. X-ray flux from sun	3 in. x 3 in. x 6 in.	3	1	
7. γ-ray spectrometer plus electronics	200 cu. in.	14	5	
8. Faraday cup	36 in. x 10 in. x 10 in.	20	4	

TABLE 1.4.2-5 (Cont'd)

MARS 1975

Parameter or Experiment	Size	Wt lbs	Power Watts	Inst. No.
I. Landers During Entry or Descent				
1. Temperature	1 in. dia. x 3 in. long	.3	.07	(I-24)
2. Pressure	1.6 in. dia. x 2 in. long	.3	.1	(I-17)
3. Density		1.5	2	(I-20)
4. Gas Chromatograph	5 in. x 5 in. x 8 in.	7	4.5	(I-8)
5. Radar Altimeter	5 in. x10 in. x 10 in. + 12 in. antenna	15	25	(I-5)
6. UV Solar Spectrometer	9 in. x 10 in. x 20 in. optics 6 in. x 10 in. x 6 in. elec.	22	12	(I-81)
7. Aerosol Profile		3	2	(I-99)
8. Electron Density (Langmuir Probe)	6 in. x 6 in. x 3 in. elec. 1 in. dia. x 6 in. long probe in freestream	3	3	(I-39)
9. Solar 3-channel Radiometer		1.5	1	(I-29)
II. Landers on Surface				
1. Temperature	1 in. dia. x 3 in. long	.3	.07	(I-24)
2. Pressure	1.6 in. dia. x 2 in. long	.3	.10	(I-17)
3. Density		1.5	2	(I-20)
4. Gas Chromatograph	5 in. x 5 in. x 8 in.	7	4.5	(I-8)
5. Wind Speed & Direction	5 in. dia. x 6 in. long	2	.5	(I-67)
6. Television - 2 cameras (1 panoramic & 1 microscope)		20 (10 ea.)	40 (20 ea.)	
7. Surface Sounds	2 in. dia. x 3 in. long includes electronics	1	1	(I-34)
8. Polarimeter - Skylight Analyzer	6 in. dia. x 3 in. long	4.5	4.5	(I-68)
9. Surface Penetrability	1 in. x 2 in. x 6 in.	1	.1	(I-25)
10. X-ray Diffractometer	10 in. x 10 in. x 10 in.	10	15	(I-32)
11. Thermal Diffusivity of Ground	3 in. x 3 in. x 3 in.	1	25	(I-64)
12. Electrical Conductivity of Ground	3 in. x 3 in. x 3 in.	1	1	(I-65)
13. Microscopic Analysis a. Atmospheric aerosols b. Surface material c. Biological materials	12 in. x 12 in. x 5 in.	15 (excluding TV #2)	7	(I-71)
14. Surface Radioactivity	200 $in.^3$	8	2	(I-13)
15. Insolation - Pyrheliometer	3 in. x 3 in. x 3 in.	1	1	(I-16)
16. Laser-induced Gaseous Emission Spectra	20 in. x 18 in. x 22 in.	50	2	(I-100)
17. Laser Atmospheric Backscatter Probe	10 in. x 10 in. x 8 in.	20	15*	(I-101)
18. Sample Handling Equipment		20		

* Intermittent Operation

TABLE 1.4.2-6: EXPERIMENTS FOR VOYAGER MISSIONS TO VENUS FOR THE OPPORTUNITIES OF 1970 AND 1972.

VENUS 1970

Parameter or Experiment	Size	Wt lbs	Power Watts	Inst. No.
I. Lander During Entry or Descent				
1. Temperature	1 in. dia. x 3 in. long	0.3	0.07	(I-24)
2. Pressure	1.6 in. dia. x 2 in. long	0.3	0.1	(I-17)
3. Density		1.5	2	(I-24)
4. Cloud Properties	50 in.^{3}	4	3	(I-41)
5. Altitude Radar Altimeter		25		
II. Lander on Surface				
1. Temperature	1 in. dia. x 3 in. long	0.3	0.07	(I-24)
2. Pressure	1.6 in. dia. x 2 in. long	0.3	0.1	(I-17)
3. Density		1.5	2	(I-24)
4. Surface Sounds	2 in. x 2 in. x 4 in.	1	1	(I-34)
5. Television Panorama		16		
6. Television Light Source		2	1	
7. Atmospheric Composition Gas Chromatograph		7	4.5	(I-8)
8. Wind Speed	5 in. dia. x 6 in. long	2	0.5	(I-67)
9. Light Levels	2 in. x 2 in. x 2 in.	0.3	0.1	(I-84)
10. Polarimeter	6 in. dia. x 3 in. long	4.5	4.5	(I-68)
III. Orbiter				
1. IR Multichannel Radiometer	5 in. x 7 in. x 4 in.	3	3	(I-2)
2. Solar Multichannel Radiometer	5 in. x 7 in. x 4 in.	3	3	(I-79)
3. Magnetic Field	4 in. x 4 in. x 6 in.	5	5	(I-23)
4. Charged Particle Flux (Ionization Chamber and G-M tube assembly)	6 in. dia. x 12 in. long	5.5	1	(I-12)
5. Televison		10.5	23	
6. Cosmic Dust	5 in. x 5 in. x 5 in.	2.5	0.2	(I-37)
7. Ionospheric Profile Bi-static Radar	4 in. x 4 in. x 12 in.	13	2	(I-85)
8. Radar Map		145	360*	

* Operating - 360 ------ Standby - 50

TABLE 1.4.2-6 (Cont'd)

VENUS 1972

Parameter or Experiment	Size	Wt lbs	Power Watts	Inst. No.
I. Lander During Entry or Descent				
1. Temperature	1 in. dia. x 3 in. long	0.3	0.07	(I-24)
2. Pressure	1.6 in. dia. x 2 in. long	0.3	0.1	(I-17)
3. Density		1.5	2	(I-20)
4. Gas Chromatrograph	5 in. x 5 in. x 8 in.	7	4.5	(I-8)
5. Altitude-Radar Altimeter	5 in. x 10 in. x 10 in. + antenna (12 in. dish)	15	25	(I-5)
6. Cloud Properties a. Liquid particles b. Solid particles 1) filter	6 in. x 6 in. x 6 in. plus 3 in.^{3} probe	7	5	(I-99)
2) light transmission	50 in.^{3} probe			(I-41)
7. Television (See Lander on Surface		11	22.2	
8. UV Solar Spectrometer	9 in. x 10 in. x 20 in. optics 6 in. x 10 in. x 6 in. elec.	22	12	(I-81)
II. Lander on Surface				
1. Temperature	1 in. dia. x 3 in. long	0.3	0.07	(I-24)
2. Pressure	1.6 in. dia. x 2 in. long	0.3	0.1	(I-17)
3. Density		1.5	2	(I-20)
4. Gas Chromatograph	5 in. x 5 in. x 8 in.	7	4.5	(I-8)
5. Wind Speed & Direction	5 in. dia. x 6 in. long	2	0.5	(I-67)
6. Television - Panorama		31.25	28.4	
7. Surface Sounds modified	2 in. dia. x 3 in. long	1	1	(I-34)
8. Skylight Analyzer	6 in. dia. x 3 in. long	4.5	4.5	(I-68)
9. Surface Penetrability	1 in. x 2 in. x 6 in.	1	1	(I-25)
10. Seismic Activity (natural, one-axis) Seismograph	5 in. dia. x 6 in. long	8	1	(I-21)
11. X-ray Diffractometer	10 in. x 10 in. x 10 in.	10	15	(I-32)
12. γ- particle scattering	3 in. x 3 in. x 3 in. sensor + 5 in. x 7 in. x 7 in. elec. (includes storage)	10	4	(I-57)
13. Thermal Diffusivity	3 in. x 3 in. x 3 in.	1	25	(I-64)
14. Electrical conductivity of Ground	3 in. x 3 in. x 3 in.	1	1	(I-65)
15. Microscopic Analysis a. atmospheric aerosols b. surface materials c. biological materials	12 in. x 12 in. x 5 in.	15	7	(I-71)
16. Insolation - Pyrheliometer	3 in. x 3 in. x 3 in.	1	1	(I-16)
17. Surface Radioactivity γ- ray spectrometer	4 in. dia. x 10 in. long + 6 in. x 6 in. x 6 in. analyzer	8	22	(I-13)
18. Meteor Trails Use the available radar - add	6 in. x 6 in. x 6 in. elec.	2.5	2.5	
19. Ionospheric Profile - Bottomside Sounder	12 in. x 12 in. x 12 in. plus antenna	70	25	(I-87)
20. Sferics	4 in. x 5 in. x 6 in. plus whip antenna	3	2	(I-82)

TABLE 1.4.2-6 (Cont'd)

III. Orbiter

Parameter or Experiment	Size	Wt lbs	Power Watts	Inst. No.
1. Magnetic Field (must be on boom)	4" x 4" x 6"	5	5	(I-23)
2. Electron Spectra and Direction (electron spectrometer)	8" x 8" x 8" includes storage & PHA	7	1.5	(I-10)
3. Proton Spectra and Direction (proton spectrometer)	8" x 8" x 8" includes storage & PHA	7	1.5	(I-11)
4. Micrometeoroids	10" x 10" x 6"	8	0.5	(I-55)
5. IR Multichannel Radio-meter	5" x 7" x 4"	3	3	(I-2)
6. Airglow Analyzer	5" x 7" x 4"	3	3	(I-93)
7. Television (color filter and zoom)		15	25	
8. Ionospheric Profile (bistatic radar)	4" x 4" x 12"	13	2	(I-85)

1.5 LANDING SITE SELECTION

The criteria which have been used in selecting the best location for the landing of the first Voyager spacecraft on the surface of Mars are the following:

1.5.1 LATITUDE

The southern hemisphere summer occurs during the period in which the planet is nearest the sun and the southern hemisphere winter occurs when the planet is at its greatest distance from the sun. The climate is, therefore, quite different in the two hemispheres, the southern hemisphere having relatively short hot summers and long cold winters and the northern hemisphere having long cool summers and short, relatively warm winters.

The 1969 spacecraft will arrive at Mars just before the winter solstice of the northern hemisphere, which of course is just before the summer solstice of the southern hemisphere. Thus biological activity in middle and high latitudes of the northern hemisphere would be at a minimum at the time of arrival of the 1969 spacecraft. However, any organic life in the southern hemisphere would be at its maximum activity at arrival time. Activity in equatorial areas would not be subject to much seasonal variation, since seasonal temperature and insolation changes are minimal at low latitudes.

1.5.2 MARIA VERSUS DESERT AREAS

The seasonal changes of the reflectance of the maria have been attributed to a seasonal change of plant life. No such reflectance changes are observed for desert areas, the implication being either that plant life is less abundant in the light-colored areas or that it is not subject to seasonal variations to the same extent as that of the maria. An additional bit of evidence for organic materials in maria is the existence of absorption at specific organic absorption bands in the 3-4 micron region of the spectrum. These bands, first detected by Sinton, are absent in radiation from the deserts.

Dr. Orr Reynolds has pointed out that most of the biological collecting and processing methods which have been developed are most suitable for application on bare soil, such as is predominant in the desert regions of the Earth. Emphasis should perhaps be shifted to the development of more versatile sample-handling methods.

1.5.3 TEMPERATURE

The lower reflectance of the maria means that a greater percentage of the solar energy incident on the planetary surface would be absorbed there. Presumably, therefore, the temperature of the dark regions would be higher than that of lighter colored regions. This is verified by observations of infrared radiant emission from the various types of areas, such data indicating that maria are about 8°C warmer than adjacent deserts.

1.5.4 TOPOGRAPHY

Topographic features of Mars are not at all well known, but Dr. Clyde Tombaugh, New Mexico State College, has concluded, after observing Mars for three decades, that there are great changes of elevation of the surface. For instance, he believes the sharp boundaries between maria and deserts to be escarpments, and that, in the mean, the maria are at a lower elevation than are the deserts.

1.5.5 SIZE OF AREA VERSUS GUIDANCE CAPABILITY

Obviously, the selected areas must be large enough so there is a high probability of being able to land the vehicles in those areas.

The two areas of Mars which have been selected for the two Landers of the 1969 Mars mission are Syrtis Major (10°N Lat., 285° Long.) and Pandorae Fretum (24°S. Lat., 310° Long.) The appearance of Syrtis Major does not change much with season, the boundaries are sharp and stable, and it is one of the darkest areas of the planet. Pandorae Fretum, on the other hand, changes considerably with season, the dark color developing in spring, deepening with the approach of summer, becoming light again in fall, and remaining light in winter.

From many standpoints it would be desirable to land in higher latitude regions. The polar caps never extend down as far as Pandorae Fretum, so close examination of the white material of the caps, probably frost, is denied. Similarly the "dark wave" will not be investigated by the Landers. However, in view of the high priority of life detection and the eventual requirement for choosing sites for manned missions, the lower latitude sites seem logical choices for the first opportunity. The landing locations for the later launches will be determined from the results of the 1969 mission.

Lack of information on the surface characteristics and rotation of Venus makes choice of landing sites difficult. The radiation measurements which have been made from Earth indicated temperature gradients with "latitude" in the sense that the "poles" are cooler than "equatorial" regions. (These terms apply only if the rotational axis is normal to the orbital plane.) It is still too early, however, to say which latitude would be most logical for landing entry capsules on the planetary surface. A Mariner 1967 Venus capsule should shed some light on this problem.

1.6 SCIENTIFIC CREDIBILITY OF VOYAGER STUDY

1.6.1 CONTRIBUTIONS FROM SCIENTIFIC CONSULTANTS

An integral part of the Voyager mission analysis has been the consultation of a number of scientists who are recognized authorities in the various scientific disciplines of importance in planetary exploration. Both the scientific orientation of the program and the definition of individual experiments have been strongly flavored by contributions from those outside consultants.

The scientists consulted, most of whom participated in the special scientific review of the Voyager study results, are listed as follows:

Dr. Ralph Alpher, General Electric Research Laboratory, Schenectady, N. Y.

Dr. Alpher worked with Dr. George Gamov on cosmological problems and has advanced, through a number of scientific publications, the theory of the evolution of the universe. His contribution to Voyager has been in detailing two cosmological experiments which can be performed aboard the spacecraft.

Dr. Dirk Brouwer, Director, Yale University Observatory, New Haven, Conn.

Because of his long career and numerous publications, including text books as well as scientific papers, Dr. Brouwer is pre-eminent in the field of celestial mechanics. His contribution to Voyager has been in defining methods of orbit perturbation analysis for planeto-physical determinations.

Dr. J. Lederberg
Dr. Elie Shneour } Exo-biologists at Stanford University
Dr. Levinthal

This team was most cooperative in discussing at length the biological problems and experiments for the detection of life on Mars. All of the specifications for their very versatile experiment, the Multivator, were obtained directly from their laboratory, with Dr. Shneour participating in the final scientific review of the Voyager experiments.

Dr. George H. Millman, General Electric Heavy Military Department, Syracuse, N. Y.

The well-known work on radar reflections from the moon and his extensive publications on incoherent scattering of microwaves in the atmosphere indicate Dr. Millman's competence in the analysis of radar problems for Voyager missions. Most of the ideas for the use of microwaves on the planets have been either contributed by or reviewed by him personally.

Dr. Richard W. Porter, General Electric Company Headquarters, New York, N. Y.

Since Dr. Porter is a member of the international Committee of Space Research and is directly involved in the work of the National Academy of Science, his review of the concepts which have resulted from the Voyager study is a timely and valuable contribution.

Prof. Zdenek Sekera, Chairman, Dept. of Meteorology, University of California, Los Angeles.

Professor Sekera's international reputation in atmospheric radiation problems is indicated by his appointment as a member of the Radiation Commission of the International Association of Meteorology and Atmospheric Physics. His contribution to Voyager has been in the evaluation and interpretation of radiation measurements in the planetary atmospheres.

Dr. S. Venkateswaran, Dept. of Meteorology, University of California, Los Angeles.

The photochemistry of the upper atmosphere and other problems of aeronomy have been the subject of Dr. Venkateswaran's research work for several years, and a number of scientific publications have resulted from that work. Most of the specific bands proposed for Voyager measurements of the airglow and the far ultraviolet radiations from the sun and planets are those suggested by Dr. Venkateswaran.

1.6.2 VOYAGER SCIENTIFIC REVIEW

This scientific review was for the purpose of bringing the Voyager study before a panel of recognized scientists for criticism and scientific evaluation of the results. The agenda consisted principally of informal presentations by members of the Voyager study team on the various phases of the overall mission, and on the details of experiments proposed for the 1969, 1971, 1973, and 1975 missions to Mars and the 1970 and 1972 missions to Venus. Comments and observations by members of the Scientific Review Panel were factored into the final system definition.

The members of the Scientific Review Panel were:

Dr. Ralph Alpher, General Electric Research Laboratory, Schenectady, New York

Prof. Dirk Brouwer, Director, Yale University Observatory New Haven, Connecticut

Dr. George Millman, General Electric Company, Heavy Military Department, Syracuse, New York

Dr. Richard Porter, General Electric Headquarters, New York, New York

Dr. Orr Reynolds, Director, Bio-Science Program, NASA, Washington, D. C.

Prof. Zdenek Sekera, Chairman, Dept. of Meteorology, University of California, Los Angeles

Dr. Elie Shneour, Dept. of Genetics, Stanford University, Palo Alto, California

Dr. S. Venkateswaran, Dept. of Meteorology, University of California, Los Angeles

1.7 POSITION OF VOYAGER IN A MARS EXPLORATION PROGRAM

In order to fully assess the scientific merit of the Voyager program it is informative to compare the scientific measurements on Mars which can be obtained from the Voyager vehicles with those which could likely be obtained by Mariner vehicles and by various configurations of manned missions. Such a comparison requires two types of value judgements to be made. They are:

Judgement of the relative value of different types of measurements - biological, atmospheric, geophysical, and interplanetary environmental; and

Judgement of the degree to which each type of system will provide the data required.

In making the first of these judgements the objectives of the planetary exploration program must be reviewed. As discussed above, life detection and its characterization is of highest priority for missions to Mars, geophysical measurements are second, and atmospheric data are third. A fourth category, which is of low priority for planetary exploration, is an improved definition of the interplanetary environment. For purposes of the present analysis, the following relative values have been arbitrarily assigned to the four types of information:

Biological	45 points
Geophysical	30 points
Atmospheric	20 points
Environmental	5 points

The judgement of the extent to which the various systems will provide the required information is a difficult one, particularly since the system concepts for manned missions are still very primitive. Of course, the basic nature of a mission may automatically eliminate certain types of measurements for that mission. For instance, it is obviously impossible to make detailed morphological analyses of surface material from an all-Orbiter configuration. Conversely, a manned Lander is ideally suited to taxonomic studies of organisms, and a comprehensive determination of the magnetic fields around the planet is best done by an orbiting vehicle.

This type of analysis for the scientific exploration of Mars is summarized in the top portion of Figure 1.7-1 for five different types of programs - Mariner, Voyager, and three types of manned missions. The various scientific areas have been divided up into the individual types of information desired, and a judgement of the ability of each of the five systems to provide the data required is indicated by the length of the appropriate bar of the graph. The distance of the line representing 100 percent capability from the base line varies in accordance with the value assignments discussed above.

It is evident from the chart that the system most nearly fulfilling all requirements is the manned orbiter combined with landing probes. The manned lander is particularly adapted to the characterization of life on the surface of Mars, but falls short in measurements of the large scale geophysical properties unless an unmanned orbiter of the Voyager type is also provided. The manned orbiter without landers has severe shortcomings in all areas. The Mariner has minimal type capabilities fairly well spread between atmospheric and geophysical measurements. Mariner also has some possibility in the detection of life, but relatively little in its characterization. For a satisfactory biological exploration of Mars, Voyager is a necessity.

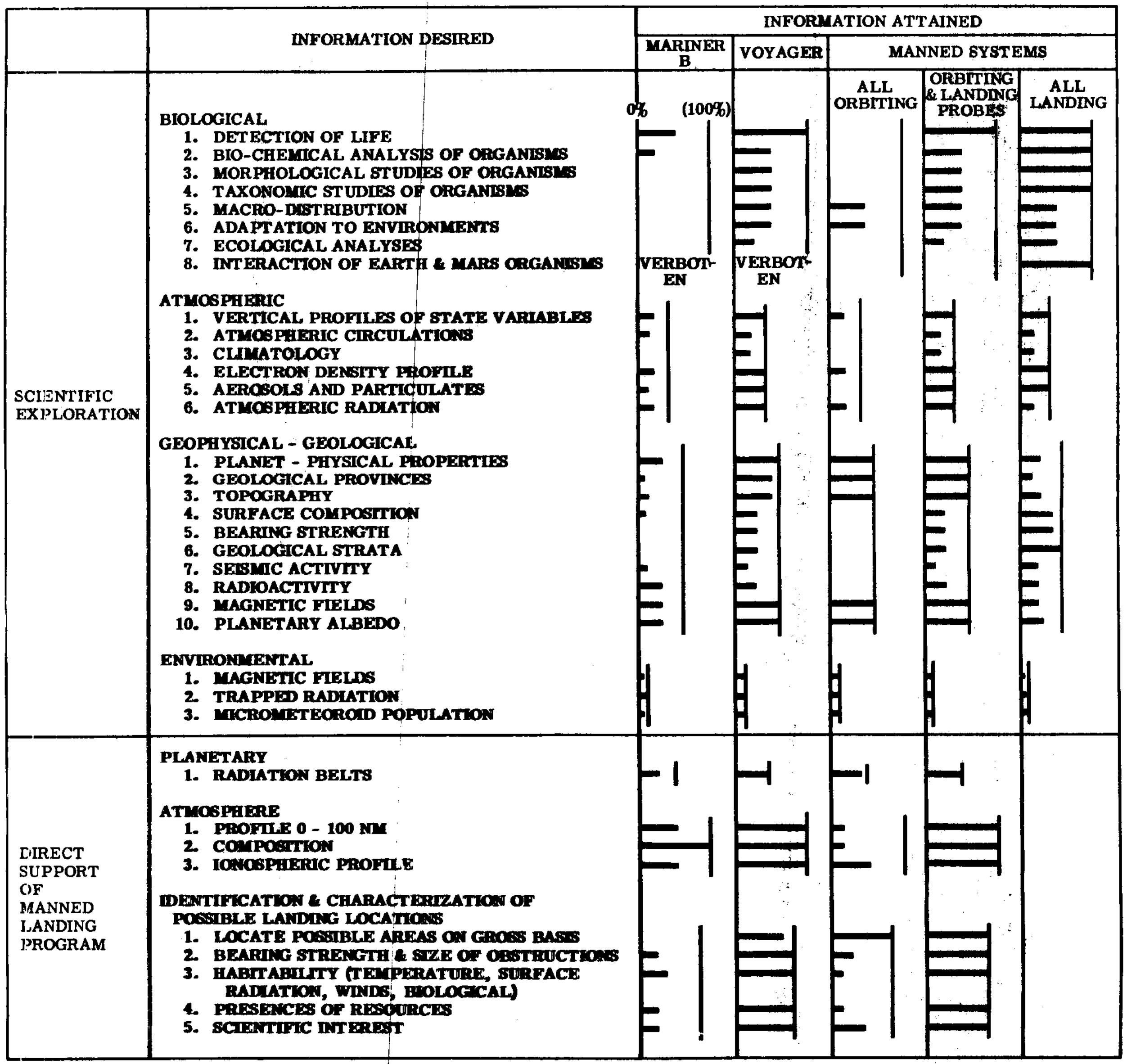


Figure 1.7.-1. Mars Exploration

Voyager is a very close second to the manned orbiter combined with landing probes in its ability to conduct a scientific exploration of Mars. (A manned lander with an unmanned orbiter would represent a much greater capability.) The orbiter-lander combinations possible on Voyager provide both versatility and excellent capability in all of the areas. However, the geological provinces and topography can probably be determined somewhat better by a manned orbiter than by the use of television alone. It should be kept in mind, however, that the cost of the manned orbiter systems is higher than that of the Voyager system by at least an order of magnitude.

In fairness to the manned systems it should be pointed out that these comparisons are based solely on the scientific measurements listed. There are undoubtedly other, more subjective, considerations which are important for manned missions but which do not show up in this type of tabulation.

A similar analysis was performed for comparing the different types of missions in furnishing the data necessary for developing manned landing systems. The results, as shown by the bottom section of Figure 1.7-1, are essentially the same as those for scientific exploration. The two most attractive systems, with little difference between them, are the manned orbiter combined with landing probes, and the Voyager system. Reasonably good atmospheric measurements can be made by Mariner, but Mariner is weak otherwise. The chart shows that the manned orbiter by itself is not well suited to supplying planetary information for the manned landing program.

In summary, it is evident that Voyager would provide a significant increase in capability over Mariner, especially in the areas of biological and geophysical-geological exploration of Mars, and provide the necessary data for the development of a manned landing system. It is the logical and necessary step between Mariner and a manned landing system.

1.8 CRITICAL DEVELOPMENT AREAS

Many of the experiments suggested for the various missions have never been utilized in space applications and are, therefore, not flight qualified for the Voyager program. The various biological experiments, some of the radiation experiments, and most of the surface analysis experiments fall in the unqualified category.

In the majority of the cases, however, the basic techniques for accomplishing the measurements are either already known or are rapidly being developed. For instance, standard techniques for gas density measurements can be adapated for measurements in the atmosphere of Mars, and gas chromatography is sufficiently flexible to encompass the required modifications for Voyager. The image orthicon camera is not yet qualified, but the very considerable efforts now under way will undoubtedly produce cameras of the required specifications for Voyager.

There are three areas, however, in which development of new techniques and equipment is critically required. They are outlined below.

1.8.1 BIOLOGICAL ASSAYS

The multiple chamber biological detector which has been developed by Dr. Lederberg's group has the versatility required for performing a number of biological experiments simultaneously if the proper assays are available. So far, according to information furnished by Dr. Shneour, only one good assay for use in the instrument has been developed. There is a critical need for further work on assays, for either that particular instrument or for other instruments. For instance, no good method is known at present for space application in detecting the presence of organisms which use light for the manufacture of food (photoautotrophs).

It is to be highly recommended that development work be begun in these critical life-detection experiments.

1.8.2 SAMPLE HANDLING AND PROCESSING

Many of the experiments proposed for Voyager are dependent on some type of sample handling. The inoculation of biological assays with organism-bearing material, the collection and transport of surface material or atmospheric aerosols to the microscope, and the use of an X-ray diffractometer all require some type of sample handling. In addition, changes of the properties of the material, such as pulverizing surface rocks, and concentration of material, such as the separation of organic and non-organic components, are very desirable, if not mandatory, for some of the experiments.

It would seem that the work presently under way on these problems is inadequate. So far, little information has been released on these activities, but it is likely that the low level of effort will not develop sufficient techniques by the time they are needed for Voyager.

1.8.3 STERILIZATION

Methods of sterilizing the various instruments are not generally satisfactory. Much more analysis of these problems is in order, from standpoints both of degradation of the instrument sensitivity and effects on reliability of instrument operation. Biological assays are particularly vulnerable to heat sterilization methods, although other types of materials are by no means immune. Further work in this whole area should be listed among the critical items for Voyager.

1.9 INSTRUMENT SPECIFICATIONS

The instruments proposed for the various Voyager missions are specified in some detail in the following pages. Each instrument is designated by an arbitrarily assigned number, preceded by the letter I. These I-numbers are used in the instrument lists given in previous tables for the specific missions.

Some of the instruments have been developed to a high degree, whereas others are still in more or less primitive stages of development. In the latter case, the specifications given should be taken as guides only.

Instrument No. (I-1)

INFRARED SPECTROMETER

Objective: To verify Sinton's measurements in the 3 - 4 micron band and detect other absorptions in the near infrared region.

Principle of Operation: A rocking prism (or grating) spectrometer is used with a suitable salt prism and infrared detector to cover the desired spectrum. In order to convert 3 decades of input into 0 to +5V output, a logarithmic amplifier will be used.

Parameter Measured: The radiation incident upon the detector will be measured as a function of wavelength in the infrared band 2u to 6u.

Dynamic Range: The spectrometer will have a dynamic range ratio of 10^3.

Response: The spectrum will be swept in less than 1 second.

Major Functional Elements:

a. Prism - A salt prism, e.g., N_a C_l, K Br, (or a grating, if used) will be rocked at the chosen rate.
b. Detector - A suitable detector, e.g., PbS, will detect the radiation.
c. Amplifier - A logarithmic amplifier will be used to furnish an output in the range of 0-5 VDC for the telemetry system.

Time of Operation: The infrared spectrometer will be used sometime after the Orbiter is in orbit. It will be controlled by both programmer and earth command.

Location: The spectrometer will be located on the planet horizontal package (PHP) and will scan in angle of elevation.

Dimensions and Power:

Weight: 29 lbs
Size: 12 in. dia. x 15 in. long
Power: 7 watts

Special Limitations: The detector may need to be cooled.

Relationship to Other Experiments: The data from the IR spectrometer will be correlated with the measurements of pressure, temperature, density, and composition of the atmosphere taken by the Lander during descent and on the ground.

Instrument No. (I-2)

INFRARED RADIOMETER

Objective: To determine the presence of H_2O vapor in the atmosphere. To determine the structure of the planetary atmosphere by making measurements of the effective radiation temperature of the planetary surface and cloud tops. To obtain the total planetary emission flux.

Principle of Operation: The planet-emitted infrared flux will be measured. Several individual radiometers will be grouped as one assembly. Each radiometer will have a specific filter and suitably matched sensor to measure a given infrared band.

Parameters Measured: The radiation incident upon the detector will be measured in the following regions:

a. 8 - 12u; - Radiation temperature.
b. 6.7u; - H_2O vapor.
c. 5 - 40u; Total planetary emission.

Dynamic Range Ratio: Each radiometer will have a dynamic range ratio of 10^3.

Response: The rise time response to a step-function input will be less than 1 second.

Major Functional Elements:

a. Filters - Filters will be chosen to pass the desired frequency bands: 8 - 12u, 5 - 40u, and a narrow band centered at 6.7u.
b. Sensors - A suitable sensor assembly will be used with each filter.
c. Amplifier - The sensor output will be converted to an instrument output in the range 0 to +5 Volts for the telemetry system.

Time of Operation: The infrared radiometer will be turned on after the Orbiter has gone into orbit. The turning on will be either by program or by Earth command.

Location: The radiometer will be mounted on the planet horizontal package (PHP) and will be aimed toward the planet.

Dimensions and Power:

Weight: 3 lbs
Size: 5 in. x 7 in. x 4 in.
Power: 3 watts

Relationship to Other Experiments: The data, tagged with time, will be transmitted to Earth. It will be correlated with other data such as altitude above planet, sun angle, longitude and latitude of the Orbiter, TV pictures of the planet's surface, and the IR spectrograph output signal.

Instrument No. (I-8)

GAS CHROMATOGRAPH

Objective:

a. To quantitatively determine the composition of planetary atmospheres.
b. To analyze solid surface material for organic constituents which may be indicative of the presence of life.

Principle of Operation: For atmospheric analyses, an accurately determined amount of gas is injected into a stream of carrier gas. The sample is transported into a partition column and an absorption column for analysis. The components in the sample are flushed through the columns at rates varying with the relative affinities of the column materials for different gases. A detector having a known sensitivity and responsivity for each component causes an imbalance in an electrical circuit which is proportional to the amount of gas present. The organic gas chromatograph will require an oven to pyrolyze material into a gaseous form. Partition and absorption columns will analyze the material for amino acids.

Parameters Measured: Gases known or believed to exist in the planetary atmosphere will be analyzed. These include part or all of the following:

H_2O (vapor), O_2, O_3, N_2, CO_2, N_2O, A, and CO. Measurements will be made to 1 ppm.

Amino acids will be analyzed so that the products from 0.1 mg of organism may be determined.

Major Functional Elements:

a. Carrier gas storage
b. Pressure regulators
c. Partition column
d. Absorption column
e. Detectors
f. Electronic circuitry
g. Pyrolysis oven
h. Sample-handling mechanism

Time of Operation: Programmed, and on command to Lander.

Location: Interior of vehicle. Intake and surface handler on exterior.

Dimensions and Power:

Weight: 7 lbs (atmospheric unit)
Size: 5 in. x 5 in. x 8 in.
Power: 4.5 watts

If a gas chromatograph capable of analyzing both atmospheric and solid constituents is to be included in a Lander, the following quantities are applicable:

Weight: 14 lbs } not including sample handler
Power: 12 watts }
Size: 8 in. x 8 in. x 10 in.

Discussion: Gas chromatographs are best utilized when the materials to be analyzed are known to be present. The design of the columns will include capabilities for analyzing all major atmospheric constituents and organic materials believed or known to be present at the time the equipment is scheduled for design.

The atmospheric and organic gas chromatographs may be included in the same package if desired. If weight constraints are imposed, either unit may be included.

Instrument No. (I-10)

CHARGED PARTICLE TELESCOPIC SPECTROMETER
Electron Spectrometer

Objective: To investigate the nature of electron fluxes about Mars.

Principle of Operation: The measurement of energy lost in passing through a thin piece of material, dE/dx, and the total energy of a charged particle can uniquely identify the particle. The use of solid state detectors, coincidence circuits, and a degree of data processing in the instrument each contribute to this identification.

Parameters Measured: The energy and flux density of electrons will be measured.

Major Functional Elements:

a. dE/dx detector - In order for the energy of any particle to be measured and telemetered, the particle must pass through the dE/dx detector.
b. Energy detector - The energy detector measures the total energy of particles which come to rest in the detector.
c. Data processor - The data processor combines the dE/dx and energy measurements, when they occur in coincidence, to automatically place the energy measurement into the electron energy data channel.

Time of Operation:

a. Continuously while in orbit.
b. Continually (twice an hour) while in transit.

Location: The electron spectrometer may be placed anywhere on the orbiting vehicle such that it has an unobstructed view of space.

Dimensions and Power:

Weight: 4 lbs
Size: 4 in. x 4 in. x 8 in.
Power: 1 watt

Discussion: The data provided by the spectrometer are related to the magnetic field strength about the vehicle. Therefore, the magnetometer and the spectrometer, as well as any other charged particle measurement, should be in operation at approximately the same time.

Instrument No. (I-11)

CHARGED PARTICLE TELESCOPIC SPECTROMETER
Proton Spectrometer

Objective: To investigate the nature of proton fluxes about Mars.

Principle of Operation: The measurement of energy lost in passing through a thin piece of material, dE/dx, and the total energy of a charged particle can uniquely identify the particle. The use of solid state detectors, coincidence circuits, and a degree of data processing in the instrument each contribute to this identification.

Parameters Measured: The energy and flux density of protons will be measured.

Major Functional Elements:

a. dE/dx detector - In order for the energy of any particle to be measured and telemetered, the particle must pass through the dE/dx detector.
b. Energy detector - The energy detector measures the total energy of particles which come to rest in the detector.
c. Data Processor - The data processor combines the dE/dx and energy measurements, when they occur in coincidence, to automatically place the energy measurements into the proton energy data channel.

Time of Operation:

a. Continuously while in orbit.
b. Continually (twice an hour) while in transit.

Location: The proton spectrometer may be placed anywhere on the orbiting vehicle such that it has an unobstructed view of space.

Dimensions and Power:

Weight: 4 lbs
Size: 4 in. x 4 in. x 8 in.
Power: 1 watt

Discussion: The data provided by the spectrometer are related to the magnetic field strength about the vehicle. Therefore, the magnetometer and the spectrometer, as well as any other charged particle measurement, should be in operation at approximately the same time.

Instrument No. (I-12)

GEIGER TUBES AND IONIZATION CHAMBER ASSEMBLY

Objective:

a. To establish whether a field of trapped radiation exists about Mars.
b. To investigate the nature of such radiation.

Principle of Operation:

a. Each ionizing event occurring within the sensitive region of a Geiger Tube results in an avalanche of electrons descending upon the centrally-located wire. This permits detection of individual ionizing particles.
b. Each ionizing particle or photon in an ionization chamber creates a quantity of ionized gas molecules and electrons which is proportional to the amount of energy left by the ionizing particle or photon inside the chamber.

Parameter Measured:

a. Each Geiger Counter will measure the flux of ionizing particles or photons which can penetrate its walls and enter its sensitive region. Wall thicknesses and wall materials may be varied from one Geiger Tube to another in order to provide gross compositional and spectral information.
b. The ionization chamber measures the amount of energy deposited within it by ionizing particles or photons which can penetrate its walls. Comparison of ionization chamber data and Geiger Tube data will add information to the gross compositional and spectral information.

Major Functional Elements:

a. Geiger counter - The counter will count individual ionizing events and will produce a series of identical pulses within the limitations of a dead-time and recovery period in which either no pulse or a modified pulse is formed. Geiger counters can count typically up to 20,000 per second.
b. Ionization chamber - The ionization chamber will produce a direct current which is changed by an auxiliary current integrating circuit into a series of of identical pulses. Each output pulse will indicate that a specific amount of energy has been deposited in the chamber and the pulse rate will provide power information, usually calibrated in terms of ergs deposited per gram of material.
c. Signal conditioning circuits - The conditioning circuits will convert each pulse from the Geiger Tubes and ionization chamber to a standard square pulse.
d. Power supply - The power supply generates the high voltage required for the Geiger Tubes and ionization chamber.

Time of Operation: Continuously while in orbit.

Location: The assembly should be located on the vehicle with as little material as possible shielding it from space.

Dimensions and Power:

Weight:	5.5 lbs
Size:	6 in. diameter x 12 in.
Power:	1 watt

Discussion: The output signals from this assembly are digital in nature and in order to prepare for a wide range of count rates, a digital accumulator should be used. By doing this, every pulse can be recorded and telemetered. In addition, it would be extremely desirable to convert each series of randomly occurring pulses to an analog rate output if it is possible to fill completely the storage volume of the accumulator.

Instrument No. (I-13)

GAMMA RAY SPECTROMETER

Objective: To perform a gross composition analysis for naturally radioactive materials on the Martian surface.

Principle of Operation: Gamma rays are detected by a scintillator - multiplier phototube assembly and the spectrum is analyzed by a pulse height analysis circuit. Spectra-peeling techniques are then used on Earth to sort out the amounts and kinds of naturally - radioactive elements producing the measured spectrum.

Parameter Measured: The gamma ray flux in each of several energy ranges is measured.

Major Functional Elements:

a. Detector Assembly - The detector assembly, consisting of a scintillating crystal, such as thallium activated cesium iodide, and a multiplier phototube, detects a flux of gamma rays by the scintillations produced.
b. Pulse Height Analysis Circuit - The magnitudes of the electronic pulses from the multiplier phototube are examined and each is determined to lie within one of several ranges. By calibration of the system these ranges can be stated in terms of gamma ray energy deposited in the scintillator.

c. Power Supply - The power supply provides the high voltage required by the multiplier phototube.

Time of Operation: After landing, for a total accumulated data producing operating time of several hours.

Location: During operation, the gamma ray spectrometer should be located in such a manner that it has an unobstructed view of the Martian surface. It does not have to lie upon or touch the surface to make its measurements.

Dimensions and Power:

Weight:	8 lbs
Size:	4 in diameter x 10 in
Power:	2 watts

Discussion: The gamma ray spectrometer will be seeking very small amounts of gamma ray activity and if a gamma source is carried on the vehicle it very likely would mask naturally-occurring radiation. Thus, the performance of this experiment requires that no gamma emitters be carried, unless their maximum gamma ray energy is less than the lowest energy of interest in the survey of surface radioactivity. The results may also be spoiled if there is on the Lander a source of neutrons which will induce gamma activity of the surface and thereby add an unknown gamma-ray background.

Instrument No. (I-16)

SOLAR RADIOMETER (PYRHELIOMETER)

Objective: To measure the total solar radiation reaching the surface of the planet.

Principle of Operation: The pyrheliometer consists of two areas, light and dark, unequally heated by solar radiation. Thermocouples connected in series electrically have their cold junctions in thermal contact with the light area and their hot junctions in thermal contact with the dark area. The electromotive force developed is thus a function of the incident solar energy.

Parameters Measured: The incident solar radiation is measured.

Dynamic Range: The pyrheliometer will have a dynamic range of 10. It will measure solar fluxes from 0.006 watts cm^{-2} to 0.06 watts cm^{-2}.

Response: The rise time response to a step function input will be less than 10 minutes.

Major Functional Elements:

a. Thermocouples - Fifty thermocouples are used in series to provide an electromotive force of about 0.1 volt for each watt cm^{-2} of solar energy input.
b. Energy absorbing areas - The light and dark areas will reflect and absorb, respectively, solar radiation in the range 0.3 micron to 2.5 micron.
c. Amplifier - A solid-state chopper stabilized d-c amplifier will be used to produce an output in the range 0 to +5 volts for the telemetry system.

Time of Operation: The solar radiometer will be used during daylight hours.

Location: The solar radiometer will be mounted high up on the Lander so that it views the sky hemisphere.

Dimensions and Power:

Weight: 1 lb.
Size: 3 in. x 3 in. x 3 in.
Power: 1 watt

Special Limitations: The solar radiometer must not be in shadow of any sort. It must accept radiation from 2π steradian. It must be pointed normal to the planet's surface.

Interpretation of Data: The incident solar radiation is given directly by means of a single constant of calibration. The data, tagged with time, will be transmitted to Earth. The sun angle will be known as a function of time, and so the incident solar radiation can be correlated with sun angle.

Relationship to Other Experiments: These data will be correlated with other data, e.g., atmospheric pressure and composition; cloud structure (as seen by the TV).

Instrument No. (I-17)

PRESSURE DETECTOR

Objective: To study the Martian atmospheric pressure.

Principle of Operation: The deflection of an elastic membrane or diaphragm, covering an evacuated chamber, is measured by an unbonded strain gauge. The strain gauge output is then amplified, processed and telemetered.

Parameter Measured:

a. The pressure detector will measure the total pressure of the Martian atmosphere at the altitude at which the measurement is made.
b. The range of pressure measurement will be from 0 to 200 mb or 0 to 50 mb.
c. The accuracy of the measurement will be $\pm$ 2% of full range.

Major Functional Elements:

a. Pressure sensing diaphragm - The diaphragm, mounted upon an evacuated (or constant pressure) chamber, changes shape as a result of changes in pressure difference between the two sides of the diaphragm.
b. Unbonded strain gauge - The strain gauge senses the change in shape of the diaphragm and reflects this change in a voltage produced by the bridge circuit of which it is a part.

Time of Operation: Continuously during descent.

Location: The pressure detector may be located anywhere within the Lander and it must have access to the atmosphere in such a manner that it may measure the atmospheric pressure (conductance between atmosphere and sensor should be high).

Dimensions and Power:

Weight: 0.3 lbs
Size: 2 in. diameter x 2 in.
Power: 1 watt

Instrument No. (I-20)

ATMOSPHERIC DENSITOMETER

Objective: To study the Martian atmosphere

Principle of Operation: The degree to which gamma rays are scattered in a gas is dependent upon both the atomic or molecular density and the atomic number(s) of the atoms. If the composition of the gas is known, the degree of gamma ray scattering may be related uniquely to its mass density.

Parameter Measured:

a. The density of the Martian atmosphere will be measured at several altitudes above the planet's surface.
b. The range of density measurement will be approximately 10^{-7} to 10^{-4} gram cm^{-3} (assuming Martian atmosphere to be similar to Earth atmosphere in-so-far as gamma scattering is concerned).
c. The accuracy of the measurement will depend upon the rate of descent of the vehicle through the atmosphere and the gamma emitter strength (curiage) but it probably can be better than $\pm$ 5% at the greater densities (degrading to $\pm$ 100% or more toward low density end of the range).

Major Functional Elements:

a. Gamma ray sources - The gamma source provides the gamma rays necessary for the scattering process.
b. Radiation shield - The shield restricts the fields of view of both the gamma ray source and the detector of scattered gamma rays. The density of the atmosphere is measured in the region in which these two fields of view overlap.
c. Detector assembly - A scintillating material, such as thallium activated cesium iodide, is used to detect the number of scattered gamma rays. The light flashes are detected and converted to electronic pulses by a multiplier phototube.
d. Power supply - The power supply provides the stable high voltage required by the multiplier phototube.

Time of Operation: Continuously during descent.

Location: The atmospheric densitometer must have an unobstructed field of view of the region in which it is to make the density measurement. Placement on the Lander aft surface would allow this requirement to be satisfied.

Dimensions and Power:

Weight: 5 lbs
Size: 2 in. x 2 in. x 6 in.
Power: 2 watts

Instrument No. (I-21)

SEISMOMETER (1-axis)

Objective: To obtain the microseismic activity of a planet. This activity is an indication of the thermal state of the planet and its tectonic activity.

Principle of Operation: A mass-magnet is suspended by a coil spring and is radially restrained so that it moves only parallel to its axis. The magnet has a circular gap which accepts a multiturn coil suspended from the frame of the instrument. The relative motion of the magnet with respect to the frame causes the magnetic field to be cut by the coil windings, thus generating an electrical signal which is proportional to the velocity of the relative motion.

Parameter Measured: The velocity of the planetary surface is measured in one direction.

Dynamic Range Ratio: The dynamic range ratio will be 10^3.

Response: The response is to be at least 20 cycles per second.

Major Functional Elements:

a. Mass-magnet assembly — This is the inertial element and electric coil assembly that comprise the sensor itself.
b. Suspension - A coil spring to suspend the mass-magnet. Natural period about 5 seconds.
c. Amplifier - The output is analog in nature and has a 30 db range. Therefore, the amplifier is logarithmic and furnishes a 0 to +5 VDC signal to the telemetry system.

Time of Operation: The seismometer will be turned on when the Lander has been oriented upon the planetary surface. It will operate continuously, since the frequency of the occurrence of tremors is unknown.

Location: The seismometer is located on the ground outside the vehicle.

Dimensions and Weight:

Weight:	8 lbs
Size:	5 in. diameter x 6 in. long
Power:	1 watt

Special Requirements:

a. Lifetime of several months is needed.
b. Axis of the instrument must not deviate more than 15° from the vertical.

Instrument No. (I-23)

MAGNETOMETER

Objective:

a. To investigate the nature of the Martian magnetic field.
b. To measure the magnitude of the Martian magnetic field.

Principle of Operation: Small rods of a ferromagnetic material which has high permeability and saturation at low magnetizing field strengths are driven into saturation by an imposed magnetic field varying sinusoidally with time. The presence of a constant external field causes an otherwise symmetrical time variation of the magnetization of the rods to become assymmetrical. The degree of assymmetry is observed in terms of harmonic content in the voltage generated in secondary windings on the ferromagnetic rods.

Parameter Measured:

a. The total magnetic field strength in the vicinity of the magnetometer will be measured by measuring and telemetering each of three orthogonal components of the magnetic field.
b. The range of measurement will be 2 to 200 gamma.

Major Functional Elements:

a. Sensors - Each component of magnetic field strength is measured by a set of two ferromagnetic cores, each of which has both a primary and a secondary winding. The windings are connected in such a manner that the fundamental frequency is cancelled and even harmonics are presented to an a-c amplifier. When no external field is present, no even harmonics are present.
b. Amplifier-Rectifier Circuits - Subsequent circuits filter, amplify, and rectify the second harmonic to provide a d-c signal uniquely related to the measured field strength.
c. Oscillator - The oscillator provides the sinusoidal currents which alternately magnetize the ferromagnetic rods first in one direction and then in the opposite direction.

Time of Operation:

a. Continually (twice an hour) during cruise.
b. Continuously during orbit.

Location: The magnetometer must be located such that it does not measure magnetic fields associated with the vehicle. This will probably require it to be mounted on a boom away from the rest of the vehicle.

Dimensions and Power:

Weight:	5 lbs
Size:	3 in. x 4 in. x 4 in.
Power:	6 watts

Instrument No. (I-24)

TEMPERATURE DETECTOR

Objective: To study the Martian atmospheric temperature.

Principle of Operation: A material whose resistance-temperature characteristics are well established is employed. The changes in resistance which this material undergoes are employed to measure temperature.

Parameter Measured:

a. The temperature detector will measure the temperature of the Martian atmosphere.
b. The range of measurement will be from $150^{o}K$ to $300^{o}K$.
c. The accuracy of the measurement will be $\pm 1^{o}K$

Major Functional Elements: Sensor - The sensing element will be a resistance thermometer whose resistance at several temperatures has been carefully established.

Time of Operation: Continuously during descent.

Location: The detector should be exposed to the atmosphere and free of any significant obstructions which may hinder motion of the atmosphere to it and thereby probably change the temperature of the gases reaching the detector.

Dimensions and Power:

Weight:	0.3 lb
Size:	1 in. dia. x 3 in.
Power:	0.1 watt

Discussion: The temperature detector described here has been considered in terms of a resistance thermometer, that is, a length of fine wire or piece of material whose ohmic resistance is employed to measure temperature. If the range of temperature measurement were to be reduced appreciably, perhaps to a range of $250^{o}K$ to $300^{o}K$, a thermistor would be a more suitable sensor and could provide greater accuracy, for such a range of measurement,than could a resistance thermometer.

Instrument No. (I-25)

PENETROMETER

Objective: To measure the soil hardness of a planetary surface.

Principle of Operation: A conical weight is dropped, point downwards, onto the soil surface. On the weight is mounted an accelerometer. The shape and magnitude of the accelerometer output will give knowledge of the hardness of the soil, whether it be quartzite or sand.

Parameter Measured: The accelerometer output is measured.

Dynamic Range Ratio: The dynamic range ratio will be 10^3.

Major Functional Elements:

a. A conical weight is mounted on a trip mechanism with the pointed end down.
b. An accelerometer is mounted to the top of the weight to measure vertical acceleration.
c. An amplifier is used to convert the output to fall in the range 0 to +5 vdc.
d. A mechanism is used to raise the weight and move it a small horizontal distance and repeat the measurement.

Time of Operation: The penetrometer will be used during the second day after the Lander has been oriented. The time of operation may be changed by Earth command.

Location: The penetrometer will be located on the Lander in such a way that it will be lowered to the planetary surface and stand within 10^{o} of the local vertical.

Dimensions and Power:

Weight:	1 lb
Size:	1 in. x 2 in. x 6 in.
Power:	0.1 watt

Relationship to Other Experiments: The data from the penetrometer will be correlated with those data from the other soil analyzer experiments and with the gravitometer.

Instrument No. (I-29)

SOLAR 3-CHANNEL RADIOMETER

Objective: To obtain data on the ionospheric density by measuring dayglow emission.

Principle of Operation: During an entry capsule's descent through a planetary atmosphere, the dayglow emission will be measured at suitable wavelengths. A three-channel radiometer assembly will be used, each channel with an appropriate narrow band filter. The assembly will scan in zenith angle through the sun, and be protected from viewing the sun itself.

Parameter Measured: The radiation incident upon the detector will be measured during the Lander's descent through the planetary atmosphere.

Dynamic Range Ratio: The dynamic range ratio will be 10^3.

Response: The rise time response to a step-function input will be less than 0.05 sec.

Major Functional Elements:

a. Filters - Filters will be narrow band and will transmit the following wavelengths:

6300 Å
6363 Å
5577 Å

b. Sensors - A multiplier phototube will be used.
c. Amplifier - Amplification will be used to produce a voltage output in the range 0 to +5 vdc.

Time of Operation: ~~The radiometer~~ will operate during descent.

Location: The radiometer will be located on the surface of the Lander in such a manner that it may scan 180° in zenith angle in a plane containing the sun.

Dimensions and Power:

Weight:	1.5 lb
Size:	3 in. dia. x 5 in. long
Power:	1 watt

Special Limitations: The radiometer must be thermally protected during entry. Entry must be during daylight.

Relationship to Other Experiments: The data, tagged with time and angle information, will be transmitted to Earth. Other data, e.g., altitude, pressure, temperature, density, will also be known versus time. Therefore, the radiometer data may be correlated with the other parameters.

Instrument No. (I-30)

NEUTRON ACTIVATION SOIL ANALYZER

Objective: To measure the chemical composition of the planetary crust material.

Principle of Operation: Neutron bombardment of a sample produces radioactive nuclides in the target material. Subsequent radioactive decay of these nuclides is generally accompanied by the emission of γ and β rays having energy spectra characteristic of those nuclides. Measurement of this spectrum permits, in principle, the identifcation both qualitative and quantitative of the elements comprising the original target material.

Practical Considerations: A few of the factors which have significant bearing on the sensitivity and precision of the method are: irradiation time, decay time, competing reactions, half-life of the product, energy of γ or β rays used for identification, interfering γ rays from products of no interest, external scattering into the detector, geometry (detector-target), self-shielding in sample inhomogeneity in flux. For instance, irradiation time for optimum activity can range from a few seconds to several hours with a source of moderate neutron flux. A similar condition exists with the decay time. It should also be pointed out that with a scintillation detector normally used, a mono-energetic flux of γ rays produces a continuous pulse height spectrum characteristic of the γ ray in question. For a polyenergetic γ ray flux which will normally exist in a space application, the pulse height spectrum is a summation of spectra of mono-energetic components. The more elements one seeks the more complex is the resulting spectrum making the interpretation a challenging task.

Parameter Measured: The γ-ray energy spectrum of the neutron-induced radioactivity of the sample is measured.

Range of Mass Numbers: Mass numbers from about 16 to 60 can be determined.

Major Functional Elements:

a. Neutron source - A neutron generator furnishing 10^7 - 10^{10} neutron sec^{-1} in the energy range of 14 Mev is used as the neutron source.
b. γ -ray counters - NaI(Tl) and plastic scintillators may be used here in combination with a multiplier phototube.
c. High voltage power supply - A high voltage ($\sim$ 1000 vdc) power supply is needed for the multiplier phototubes.
d. Electronics - Amplifiers and multichannel pulse height analyzers are needed to store the spectral information.

Time of Operation: The analyzer will be put into operation about 3 days after the Lander has been oriented. The time may be changed by Earth command.

Location: The analyzer will be located anywhere in the Lander, since a sample will be brought to it.

Dimensions and Power:

Weight:	20 lbs
Size:	4 in. dia. x 36 in. long detector and generator
	4 in. x 4 in. x 3 in. analyzer
Power:	20 watts

Relationship to Other Experiments: This analyzer supplements the data taken by the scattered α-particle soil analyzer and by the X-ray diffractometer.

Instrument No. (I-32)

X-RAY DIFFRACTOMETER

Objective: To identify the planetary crust by mineralogic analysis.

Principle of Operation: When X-rays illuminate a powdered mineral, the diffracted X-ray pattern is unique for that mineral. Therefore, a mineral sample is prepared by grinding a specimen of planetary crust to a powder. The specimen must be ground so that the maximum crystallite size does not exceed a value that limits the random orientation of powder components, nor so small that pattern is unrecognizable. The radiation source is a miniature X-ray tube. The proportional Geiger counter detector is mounted in a goniometer which in turn is geared to the sample holder in such a manner that the angle relationship of Bragg's law is satisfied.

Parameter Measured: The X-ray diffraction pattern is measured.

Dynamic Range: 30 to 3500 counts sec $^{-1}$.

Resolution: The width of the ionization peak, at 1/2 max. height, to be no greater than $2\theta = 0.2$ degree, and at 0.1 height not to exceed 0.4°. Total scan region to cover 7° to 180° at rates of 0.5 and 4 degrees min $^{-1}$.

Major Functional Elements:

a. X-ray source - A miniature X-ray tube operating at 25 kw and having a Cu anode is used as the source of radiation.
b. Detector - A proportional Geiger counter is used for the detector. The detector rotates about an axis located at the sample holder.
c. Electronics - The electronics contains the high voltage power supplies, amplifiers and storage for 50,000 bits.

Time of Operation: The X-ray diffractometer will be activated 4 days after landing on Mars and immediately after landing on Venus.

Location: The diffractometer may be located anywhere in the Lander since the prepared sample is brough to it.

Dimensions and Power:

Weight:	10 lbs
Size:	10 in. x 10 in. x 10 in.
Power:	15 watts

Special Limitations: The sample material must be crushed so that crystallite sizes fall within the range of 75 Å to 500 Å .

The peak-signal-to-background noise ratio must be a minimum of 30:1.

Relationship to Other Experiments: Data from this experiment will be related to data from the neutron activation and α-scattering experiments.

Instrument No. (I-34)

MICROPHONE

Objective: To listen for sounds on Martian surface.

Principle of Operation: The sound pressure waves can be used to bend or distort slightly an electrically conductive diaphragm which is one side of a capacitor. The resulting changes in capacitance can indicate the instantaneous sound pressure acting on the diaphragm.

Parameter Measured: The microphone will measure sounds. It is intended that the telemetry will process the microphone signal in such a manner that the sounds heard on Mars can be faithfully reproduced (within certain intensity and frequency limitations) on Earth.

Major Functional Elements:

a. Microphone - The microphone will be basically a standard but rugged capacitance type microphone.

b. Signal Processor - The microphone signal will be processed by a compressor (log amplifier and/or automatic gain control circuit) in such a manner that a complementary expander circuit may be used on Earth to compensate for the deliberate signal distortion.

Time of Operation: After landing, continuously for several minutes every hour.

Location: The microphone "receiving surface" or face must be either at the vehicle's surface or outside of the vehicle.

Dimensions and Power:

Weight:	1 lb
Size:	2 in. dia. x 3 in. (microphone and preamplifier) and 2 in. x 2 in. x 2 in.(electronics)
Power:	1.0 watt

Instrument No. (I-36)

PRECIPITATION METER

Objective: To measure the amount and kind (solid or liquid) of any precipitation that may occur.

Principle of Operation: Precipitation is collected by a graduated conical collector having a length to maximum diameter ratio of 10. Any large amount of precipitation can be measured by viewing the collector directly with the TV camera. Solid precipitate will remain, and so a method of dumping the collector is provided. Liquid precipitate will evaporate, and so the cone is coated with a dye which changes color when it becomes wet. This makes liquid precipitate more readily visible in the TV system. Adjacent to the conical collector is an inclined plate coated with dye which changes color when it becomes wet. This color change is irreversible. Calibration colors are also set up in view of the TV camera. A suitable color filter wheel is provided through which the TV views the precipitation meter assembly.

Parameters Measured: The amount and color of precipitation are measured. Detection of past liquid precipitation is also provided.

Major Functional Elements:

a. Conical collector - A glass cone is used, having calibration marks on the outer surface visible in the TV camera.
b. Color filter - The TV views the collector through color filters contained in a wheel. This enables the type of precipitate to be known immediately.
c. Light - An illumination light is provided for viewing at night if it should be desirable.

Time of Operation: The precipitation meter will always be in operation so that it may be viewed at will by the TV camera.

Location: The precipitation meter is located on the antenna boom of the Lander.

Dimensions and Power:

Weight:	1 lb
Size:	10 in. long x 5 in. x 3 in.
Power:	1.0 watt light (not always used) + 1/4 watt dumping motor power

Instrument No. (I-37)

COSMIC DUST DETECTOR

Objective: To obtain direct measurements of the cosmic dust particle momentum and mass distribution near the Mars - Deimos - Phobos system.

Principle of Operation: Two sensors will measure dust particle impact by direct penetration and by microphonic techniques. The microphone is the only analytical sensor. Capacitors indicate the side of the sensor that the microphone pulse height analysis (PHA) is taken and they count the number of impacts that occurred, some of which will be below the detection threshold of the microphone.

Parameters Measured:

a. The number of impacts
b. The particle momentum

Momentum Sensitivity: 10^{-5} dyne sec.

Penetration Mass: 10^{-12} grams will be detected.

Major Functional Elements:

a. Sensor - The sensor is a stainless steel rectangular microphone plate 0.015 inches thick with a piezo-electric pick off crystal mounted to the back; it is sensitive on both sides. Both sides are coated with 2000 Å of non-conducting material and covered with an evaporated layer of aluminum; forming the penetration capacitor sensors with the microphone plate as a common side of both capacitors.
b. Amplifier - The output of the microphone ringing is fed into an amplifier having a voltage gain of 100 db and a 100 kc bandpass.
c. Data Storage - Two 8-bit storage units are used, alternately read out as 8 bits per frame.
d. Power Converter - The Power Converter supplies +50 volts for the capacitor sensors and +6 volts for the remaining electronics.

Time of Operation: The Cosmic Dust Detector will go into operation as soon as the Orbiter is put into orbit.

Location: The Cosmic Dust Detector will be located in an exposed area on the Orbiter.

Dimensions and Power:

Weight:	2.5 lbs
Size:	5 in. x 5 in. x 5 in.
Power:	0.2 watt

Reference: Mariner C Spacecraft Flight Equipment Design Specification No. MC-4-224A. JPL

Instrument No. (I-39)

LANGMUIR PROBE

Objective: To measure the electron density as a function of altitude in the planetary upper atmosphere during entry.

Principle of Operation: The Langmuir Probe consists of a small electrode, or probe, projected at the point of measurement and a large electrode, namely the entry vehicle, as a ground plane. A voltage waveform, varying in amplitude and polarity, is applied across the two electrodes and the circuit current is measured. From the experimentally determined current-voltage characteristic, the electron temperature and the electron density may be computed.

Parameters Measured: The current versus voltage characteristic of a set of electrodes is measured as a function of time.

Dynamic Range: The probe will have a dynamic range of between 2 and 4 decades, depending upon the current collected.

Response: The rise time response to a step function input will be less than 0.1 sec.

Major Functional Elements:

a. Small probe - This collects the electrons and ions at the point of measurement.
b. Large probe - The large probe, area $\approx$ 100 x area of small probe, acts as ground plane. Commonly, it is tied to the aft wall of the Lander.
c. Power supply - The power supply furnishes a sweeping voltage between the two probes.
d. Amplifier - The amplifier amplifies the probe current to furnish an instrument output for the telemetry system to sample.

Time of Operation: The Langmuir Probe will operate from the beginning of entry until it is burned off.

Location: The sensing probe will be mounted out beyond the plasma surrounding the Lander, so that it measures the ambient electron density.

Dimensions and Power:

Weight:	3 lbs
Size:	1 in. dia. x 6 in. long + 6 in. x 6 in. x 3 in. electronics
Power:	3 watts

Special Limitations: The measurements will be limited to the altitude range through which the probe survives.

Interpretation of Data: The interpretation of the data is highly dependent upon the effective sampling probe dimensions and size, relative ion and electron temperatures, size of probe relative to the mean free path of electrons at the time of measurement, electron density perturbations caused by the probe, voltage sweep rate, and varying space potentials.

Relationship to Other Experiments: The data, tagged with time, will be transmitted to Earth. These data will be correlated with other data, e.g., atmospheric composition, altitude, density, solar radiation levels versus wavelength, etc.

Instrument No. (I-40)

RADIOMETER

Objective: To obtain information about the vertical profiles of 0 and 0_2 densities.

Principle of Operation: Lyman β radiation from the sun is absorbed by O, which fluoresces at 8446 Å (Bowen mechanism). This radiometer will measure such fluorescence. It must be used in conjunction with the ultraviolet radiometer during entry into a planetary atmosphere. The radiometer will scan in zenith angle through the sun, yet not view the sun itself.

Parameter Measured: The radiation incident upon the detector will be measured during entry into a planetary atmosphere.

Dynamic Range Ratio: The dynamic range ratio will be 10^3.

Response: The rise time response to a step-function input will be less than 0.05 sec.

Major Functional Elements:

a. Filter - A narrow band filter centered at 8446 Å will be used.
b. Detector - A suitable detector will be used.
c. Amplifier - Amplification will be used to obtain an output voltage in the range of 0 to +5 vdc.

Time of Operation: The radiometer will operate during entry into a planetary atmosphere.

Location: The radiometer will be located on the surface of the Lander in such a manner that it may scan 180° in zenith angle in a plane containing the sun.

Dimensions and Power:

Weight:	1 lb
Size:	3 in. dia. x 5 in. long
Power:	1 watt

Special Limitations: Entry must be during daylight.

Relationship to Other Experiments: The data, tagged with time and angle information, will be transmitted to Earth. These data will be used with the data from the ultraviolet radiometer and radar altimeter to deduce the amounts of 0, 0_2, and N_2 in the upper atmosphere.

Instrument No. (I-41)

CLOUD ANALYZER

Objective: To determine whether or not the descending Venus Lander is passing through cloud, and to obtain vertical dimensions of clouds on Venus.

Principle of Operation: A modulated beam of light shines across the base of the Lander onto a mirror and reflects about 90° into a detector in the Lander. Ambient light also reaches the detector. As the Lander descends, any cloud formation will eddy behind the lander and change the optical transmission of the light beam. The signal in the detector will thus be in two parts: a d-c component giving ambient light levels and an a-c part from which the presence and absence of cloud may be determined and also the optical transmission of the cloud may be obtained.

Parameters Measured:

a. The transmission of a given optical path across the base of the Lander is measured.
b. The ambient light level, in the visible and near IR spectral region, is measured.

Major Functional Elements:

a. Light source - A stroboscopic light is used, well reflected and collimated and modulated at about 1000 cycles sec^{-1}.
b. Detector - A photo cell is used as a detector.
c. Mirror - A stationary mirror is used to reflect the light beam about 90^{o} into the detector.

Time of Operation: The cloud analyzer will be used during descent to Venus.

Location: The cloud analyzer is located on the aft end of the Lander.

Dimensions and Power:

Weight:	4 lbs			
Size:	Three pieces:	a.	light source	- 2 in. x 2 in. x 6 in. long
		b.	detector	- 2 in. x 2 in. x 6 in. long
		c.	mirror	- 1 in. x 1 in. x 1 in.
Power:	3 watts			

Relationship to Other Experiments: The data from this experiment will be correlated with those data from the radar altimeter, pressure, temperature, and density experiments.

Instrument No. (I-43)

MASS SPECTROMETER

Objective: To perform an analysis of the atmospheric composition.

Principle of Operation: If a mixture of different kinds of ions are accelerated so that they have either the same energy or the same momentum, they will separate while in free flight because the speed of each ion is a function of the ratio of its charge to its mass. If each ion is singly charged, the time of flight between two points is a direct indication of its mass.

Parameters Measured:

a. The spectrometer indicates the masses of the ionized molecules in the atmosphere.
b. The spectrometer measures the partial pressure of each kind of ion detected in the atmosphere.

Major Functional Elements:

a. Ion source and accelerator - Molecules from the atmosphere are led from a sample inlet to the chamber in which they are ionized, usually by electron bombardment, and then accelerated by a pulsed electric field.
b. Drift tube - The ions move through the drift tube in response to the pulsed electric field and separate into groups according to their charge to mass ratio.
c. Ion detector - The ion detector is an electron multiplier which amplifies each ion pulse.

d. Data processor - The pulses amplified by the electron multiplier tube are sorted and converted to d-c voltage levels proportional to partial pressure of each atmospheric constituent.
e. Power supply - The power supply provides the several special voltages which accelerate the ions and amplify the ion pulses.

Time of Operation:

a. Continuously during descent and immediately after landing
b. Continually during orbit

Location: The spectrometer must be mounted such that its sampling port has direct access to the space about the vehicle.

Dimensions and Power:

Weight: 6 lbs.
Size: 3 in. x 5 in. x 10 in.
Power: 6 watts

Discussion: The mass spectrometer will be able to resolve atomic mass units from 12 to 50 and obtain data with slightly less resolution to approximately 200 amu. When data from the gas chromatograph are compared with those from the mass spectrometer, the atmospheric constituents will be identified.

Instrument No. (I-57)

α-SCATTERING SOIL ANALYZER

Objective: To measure the chemical composition of the planetary crust material.

Principle of Operation: The energy spectrum of scattered particles depends on the mass numbers of nuclei within the target material and inversely on the atomic stopping powers for charged particles of the target material. For thick targets, a continuous spectrum is obtained exhibiting sharp, high energy cut-off edges whose position is unaffected by the chemical or physical state of the scatterer. For elements heavier than aluminum and low energy α particles, large angle scattering is primarily Rutherford's scattering. For lighter elements and higher energy α particles (α, ρ) reactions become important. The spectrum of the resulting protons contains information similar to that obtainable from α particle spectrum. The breakpoints in the spectrum identify the element, and heights between plateaus are a measure of relative elemental abundances.

Parameter Measured: The energy spectrum of scattered α-particles is measured.

Range of Elements Measured: Individual elements can be identified through calcium, with the exception of hydrogen. Elements of higher atomic mass can be identified only as to group.

Major Functional Elements:

a. α-particle source - The energy of the α-particles is about 6 Mev.
b. Detector - A solid state detector is used.
c. Electronics - An amplifier, pulse height analyzer, and storage is used.

Time of Operation: The analyzer will be put into operation about 3 days after the Lander has been oriented. The time may be changed by Earth command.

Location: The analyzer will be located anywhere in the Lander, since a sample will be brought to it.

Dimensions and Power:

Weight: 7 lbs
Size: 3 in. x 3 in. x 3 in. sensor plus 5 in. x 7 in. x 7 in. active electronics plus ~ 10^4 bit storage
Power: 2 watts

Relationship to Other Experiments: The data from the α-particle scattering soil analyzer will be correlated with those data from the neutron activation soil analyzer and the X-ray diffractometer.

Instrument No. (I-64)

THERMAL DIFFUSIVITY METER

Objective: To obtain the thermal diffusivity of the planetary topsoil.

Principle of Operation: A small heat sensor is placed in contact with the soil sample. External heat is applied from a black body source. From the time-temperature history of the material, the thermal diffusivity may be determined.

Parameter Measured: The temperature of the sample vs. time is measured.

Range: The range of diffusivities will be from 10^{-2} to 10^{-6} cm^2 sec^{-1}.

Major Functional Elements:

a. Heat sensor - A small thermistor is used to measure the temperature of the sample.
b. Heat source - A small black body source is used.

Time of Operation: The diffusivity meter will be used 3 days after the Lander is oriented. The time may be changed by Earth command.

Location: The diffusivity meter will be placed upon the planetary surface by the Lander.

Dimensions and Power:

Weight: 1 lb.
Size: 3 in. x 3 in. x 3 in.
Power: 25 watts

Relationship to Other Experiments: The data will be correlated with those data from other soil analysis experiments.

Instrument No. (I-65)

ELECTRICAL CONDUCTIVITY METER

Objective: To measure the electrical conductivity of the planetary surface soil.

Principle of Operation: The electrical conductivity of planetary soil can be determined by placing a resonant tank circuit near the soil and measuring the Q of the circuit.

Parameter Measured: The eddy current losses in the material are sensed.

Major Functional Elements:

a. A resonant tank circuit
b. Electronics to measure the tank circuit dissipation

Time of Operation: The electrical conductivity meter will be used once every 3 hours for 3 days beginning 12 hours after the Lander has been oriented upon the planetary surface.

Location: The electrical conductivity meter will be placed on the planetary surface by the Lander.

Dimensions and Power:

Weight: 1 lb
Size: 3 in. x 3 in. x 3 in.
Power: 1 watt

Relationship to Other Experiments: The data will be correlated with the data from the other soil analyzer experiments.

Instrument No. (I-67)

ANEMOMETER

Objective: To observe the atmospheric motion near surface of Mars.

Principle of Operation: Atmospheric movement or wind results in a force being applied to any and all objects in the path of the wind. The magnitude and direction of this force is related to the wind velocity.

Parameters Measured:

a. Wind speed
b. Wind direction

Major Functional Elements:

a. Vane - A 3-dimensional,cruciform-shaped vane is suspended in such a manner that wind may exert pressure on it from any direction.
b. Pressure transducers - The force in 3 orthogonal directions, of known orientation, is measured by pressure transducers.

Time of Operation: Continuously after landing, for several seconds to minutes, several times a day.

Location: After landing, the anemometer will be extended a short distance above the vehicle.

Dimensions and Power:

Weight: 2 lbs
Size: 5 in. diameter
Power: 0.5 watt

Instrument No. (I-68)

POLARIMETER - SKYLIGHT ANALYZER

Objective:

a. To determine the intensity and polarization of skylight.
b. To measure Rayleigh-type scattering in the near UV.
c. To measure surface reflection in the near IR.
d. To measure the integrated planetary albedo across a wide range of wavelengths.

Principle of Operation: The intensity of skylight is measured by suitable multiplier phototube sensors. There are appropriate color filters and polarizing filters introduced in front of the sensors. The sensor output is thus a function of the intensity, polarization, and wavelength of the incident skylight.

Parameters Measured: Skylight intensity incident upon the detectors is measured. The intensity is also measured with color filters and polarizing filters in the beam.

Dynamic Range Ratio: The skylight analyzer will have a dynamic range ratio of 10^3.

Response: The assembly completes a cycle of measurements in 30 seconds.

Major Functional Elements:

a. Sensors - The light sensitive sensors are side viewing multiplier phototubes.
b. Filters - A dual filter wheel assembly is used. One wheel contains three polarizing filters set with their directions of polarization at 60° with respect to one another. The other wheel contains color filters covering the following bands:

   1) 3000 Å - 4000 Å
   2) 5000 Å - 6000 Å
   3) 7500 Å - 8500 Å
   4) 0.3μ - 3.0μ
c. High voltage power supply - A high voltage power supply furnishes power for the two multiplier phototubes.
d. Amplifier - An amplifier produces an output of 0 to +5 volts for insertion into the telemetry system.
e. Shutter - An automatic shutter is used to prevent direct sunlight from ever striking the multiplier phototubes.

Time of Operation: The skylight analyzer will be turned on sometime after the Lander has been steadied. It is turned on only during daylight hours.

Location:

a. Landers - The polarimeter is located in a position such that it views the sky, preferably vertical in the first Landers destined for landings in the low latitudes.
b. Orbiters - The polarimeter is located on the planet horizontal package (PHP) and views the planet.

Dimensions and Power:

Weight: 4.5 lbs
Size: 6 in. dia. x 3 in. long
Power: 4.5 watts

Special Limitations: The orientation of the skylight analyzer must be known.

Interpretation of Data: From the intensities of the fluxes through the three polarizing filters, the polarization degree and angle may be computed. Since these data will be taken through color filters, the wavelength dependence will also become known.

Relationship to Other Experiments: The data, tagged with time, will be transmitted to Earth. The sun angle is known as a function of time, and so the polarization degree and angle may be correlated with sun angle. These data will be correlated with other data, e.g., atmospheric pressure, composition, and density.

Instrument No. (I-69)

INSECT ATTRACTOR

Objective: To aid in the search for life on Mars.

Principle of Operation: A light source will be used to attract insects in much the same manner as such sources attract insects on Earth. A material which emits an attractive odor may also be included in the device.

Parameters Measured: The insect attractor would not perform any measurements. It would act as an auxiliary piece of equipment to the television and microphone by increasing the likelihood of their detecting life.

Major Functional Elements:

a. Light source - If the behavior of insects on Earth be used as a guide, the light source will be one which emits much blue, violet and near ultraviolet light. It probably would be of little value to provide a warm light source unless warm-blooded animals are expected to exist on Mars.
b. Odor source - Again, if the behavior of insects and plants on Earth be used as a guide, a substance which emits a suitable odor might be used in the insect attractor. Such a general attractant might be a sugar-like substance. The choice of a suitable material will require careful consideration and speculation.

Time of Operation: The insect attractor will operate after the entry vehicle has landed and while the television and microphone are in operation. It should operate best during daylight and at dusk.

Location: The insect attractor will be placed in contact with the ground in order to attract crawling as well as flying insects. It will be located near the microphone and in the field of view of the television camera.

Dimensions and Power:

Weight: 0.1 lb
Size: 1 in. x 1 in. x 1.5 in.
Power: 1.0 watt

Discussion: As has been mentioned already, the insect attractor operates only as an aid to the television and microphone. The television will have to be capable of looking in the direction of the attractor and of supplying descriptive detail in objects a fraction of an inch in size on the ground and in the air about the attractor.

Instrument No. (I-70)

SOIL MOISTURE GAUGE

Objective: To obtain the amount of moisture in the soil of Mars.

Principle of Operation: The sample handling equipment dumps a known mass of soil into the moisture gauge's hopper and seals the only entrance. The thermally-insulated hopper is heated to 200°C to drive out any moisture in the soil sample. The moisture passes through a connecting tube into a cool chamber containing a moisture detector. The detector resistance is a function of moisture present in the enclosed atmosphere. From this reading, the moisture may be obtained.

Parameters Measured: The moisture in a given volume of atmosphere is measured.

Major Functional Elements:

a. Heated hopper - The soil sample is placed in the hopper which is then heated.
b. Moisture detector - The moisture in the atmosphere surrounding the detector determines the resistance of the surface of the detector.

Time of Operation: The soil moisture gauge will be activated sometime during the second night or third day. The gauge also may be turned on by Earth command as desired.

Location: The soil moisture gauge is located in the Lander, andsoil samples are brought to it.

Dimensions and Power:

Weight: 2 lbs
Size: 3 in. x 3 in. x 10 in.
Power: 25 watts intermittent

Relationship to Other Experiments: The soil moisture data will be correlated with the data from the other soil analysis experiments.

Instrument No. (I-71)

MICROSCOPE ANALYZER

Objectives:

a. To obtain the characteristics of whatever aerosols there may be in the atmosphere of Mars at the planet's surface.
b. To examine mineralogical samples of surface materials.
c. To examine the surface for biological materials.

Principle of Operation: Any solid or liquid aerosols filtering downward will be caught on a flat plate and transported to the microscope for viewing. Secondly, a filter will be exposed to the atmosphere and will have the atmosphere drawn through it. The aerosol material deposited upon the filter will be brought to view in the microscope. Samples from the planet's surface will be brought to the microscope for viewing.

Parameters Measured: The aerosols, mineralogical samples, and biological samples will be observed and measured as to size, number density, shape, and appearance.

Major Functional Elements:

a. Microscope - This includes the microscope itself, light source, color filters, polarizing filters, and actuating motors.
b. TV Camera - This includes all electronics to operate the camera which views the microscope image.
c. Sample handler - This is the portion of the overall handling equipment which moves the samples into view of the microscope.

Time of Operation: The sample handlers will operate at various times collecting samples. In general, the microscope will be operated during the night; thus leaving the shared TV telemetering system available for daylight landscape viewing.

Location: The microscope may be mounted anywhere so long as the sample handling equipment is suitable.

Dimensions and Power:

Weight: 15 lbs (not including TV camera & sample handling).
Size: 12 in. x 12 in. x 5 in.
Power: 7 watts

Instrument No. (I-72)

GRAVITOMETER

Objective: To measure the acceleration of gravity on the planetary surface at the location of the Lander.

Principle of Operation: The period of a physical pendulum is a function of the constructional geometry of the pendulum and of gravity. Therefore, knowing the construction and measuring the period will allow the local gravity to be computed.

Parameter Measured: The period of the physical pendulum will be measured.

Dynamic Range: The gravitometer will measure accelerations between 10 ft sec $^{-2}$ and 15 ft sec $^{-2}$.

Major Functional Elements:

a. Pendulum - A physical pendulum is mounted on low friction bearings.
b. Timer - An electronic timer will measure the period of the pendulum.

Time of Operation: The gravitometer will be turned on one day after landing. The time of operation may be changed by Earth command.

Location: The gravitometer may be mounted anywhere within the Lander and oriented with a given axis parallel to local gravity.

Dimension and Power:

Weight: 3 lbs
Size: 5 in. x 5 in. x 3 in.
Power: 3 watts

Instrument No. (I-78)

UV MULTICHANNEL RADIOMETER

Objective: To determine the presence of particular constituents of the upper atmosphere by means of filter spectroscopy; in particular, H_2O vapor, O, O_2, O_3, N_2.

Principle of Operation: During entry, the amount of selected portions of the solar radiation penetrating a planetary atmosphere is measured as a function of the altitude above the planetary surface. Several individual radiometers are grouped as one assembly, and the assembly is aimed at the sun. Each radiometer will have a specific filter and suitable matched sensor to observe a given portion of the spectrum.

Parameters Measured: The radiation incident upon the detector will be measured during the Lander's entry into the upper atmosphere.

Dynamic Range Ratio: The radiometer will have a dynamic range ratio of 10^3.

Response: The rise time response to a step-function input will be less than 0.25 sec.

Major Functional Elements:

a. Filters - Filters will transmit a narrow band of wavelengths centered at the following wavelengths:

   1215 Å (Lyman α)
   1026 Å (Lyman β)
   972 Å (Lyman γ)
   584 Å (HeI)
   304 Å (HeII)

   In addition, the following relativity broad bands will be defined by filters:

   1445 Å to 1500 Å
   2500 Å to 3000 Å

b. Sensors - A suitable sensor will be used with each filter.

c. Output network - The output from each channel will lie in the range 0 to +5 volts. (This will match the telemetry requirements.)

Time of Operation: The radiometer will measure the penetrating solar radiation continuously during entry.

Location: The radiometer will be located on the Lander in such a manner that it will point toward the sun during entry.

Dimensions and Power:

Weight: 1.5 lbs
Size: 3 in. dia. x 5 in. long
Power: 1.5 watts

Special Limitations: The radiometer must be thermally protected during entry. Entry must be during daylight.

Relationship to Other Experiments: The data, tagged with time, will be transmitted to Earth. These data will be correlated with other data, e. g., altitude and the data from the radiometer covering the 8446 Å line of O fluorescence. From these interrelated data, information can be obtained about the vertical profiles of O, O_2, O_3, and N_2 densities.

Instrument No. (I-79)

SOLAR MULTICHANNEL RADIOMETER

Objective: To measure Rayleigh type scattering in the near UV; to measure surface reflection in the near IR, and to measure the integrated planetary albedo in a broad band of wavelengths.

Principle of Operation: The intensity of skylight is measured by suitable multiplier phototube sensors. There are appropriate color filters in front of the sensors. The sensor output is thus a function of intensity and wavelength of the incident skylight.

Parameters Measured: Skylight incident on the radiometers is measured as a function of wavelength and time.

Dynamic Range Ratio: The dynamic range ratio will be 10^3.

Response: The rise time response to a step function input will be less than 1 second.

Major Functional Elements:

a. Sensors - The light sensitive sensors are side-viewing multiplier phototubes.

b. Filters - The filter wheel contains color filters covering the following bands:

1)	3000 Å to 4000 Å	Rayleigh-type scattering
2)	5000 Å to 6000 Å	Visible light
3)	7500 Å to 8500 Å	Surface reflection
4)	3000 Å to 30,000 Å	Planetary albedo

c. High voltage Power Supply - A high voltage power supply furnishes power for the two multiplier phototubes.

d. Amplifier - An amplifier produces an output of 0 to +5 vdc

e. Shutter - An automatic shutter is used to prevent direct sunlight from ever striking the multiplier phototube light-sensitive surface.

Time of Operation: The Solar Multichannel Analyzer will be turned on when the Orbiter is in orbit.

Location: The analyzer is located on the planet horizontal package (PHP) and views the planet.

Dimensions and Power:

Weight: 3 lbs
Size: 5 in. x 7 in. x 4 in.
Power: 3 watts

Relationship to Other Experiments: The data, tagged with time, will be transmitted to Earth. The data will be correlated with the sun angle and with data taken by the polarimeter from the surface of the planet.

Instrument No. (I-81)

UV SOLAR SPECTROMETER

Objective: To obtain vertical profiles of planetary atmospheric constituents of Mars and to obtain the atmospheric reflection properties of Venus.

Principle of Operation: For the vertical profiles, a grating spectrometer is carried by a Lander and is used viewing the sun during entry. For reflection measurements, the spectrometer is carried by an Orbiter and views the planet.

Parameters Measured: The spectral distribution of the solar radiation (transmitted or reflected) incident upon the spectrometer is measured in the wavelength range from 500 Å to 2500 Å.

Dynamic Range Ratio: The dynamic range ratio will be 10^3.

Response: The spectrum will be swept in less than 0.1 second with the spectrometer on the Lander, and in less than 1.0 second with the spectrometer on the Orbiter.

Major Functional Elements:

a. Grating and Optical Assembly - The grating and optical assembly will cover the spectral range 500 Å to 2500 Å.

b. Detector - A suitable detector will be chosen for this wavelength range.

c. Amplifier - The amplifier will furnish an output voltage in the range 0 to +5 vdc

Time of Operation: The spectrometers mounted on Landers will operate during entry. The spectrometers mounted on Orbiters will operate after the orbit has been established.

Location: The spectrometers on Landers will be located so that they view the sun during entry. The spectrometers on Orbiters will be mounted on the planet horizontal package and will view the planet:

Dimensions and Power:

Weight: 22 lbs
Size: 9 in. x 10 in. x 20 in. optics
6 in. x 10 in. x 6 in. electronics
Power: 12 watts

Special Limitations: The Landers must enter the atmosphere during daylight.

Relationship to Other Experiments: The spectral distribution will be tagged with time and transmitted to Earth. The data will be correlated with UV data from other experiments and with atmospheric composition, temperature, and density data to establish a complete set of atmospheric profiles.

Instrument No. (I-82)

SFERICS DETECTOR

Objective: To obtain data about the number of thunder storms in a planet's atmosphere.

Principle of Operation: Lightning discharges during thunderstorms are a source of radio waves popularly known as static. A broad band receiver is used with a whip antenna to monitor these sferics.

Parameters Measured: The number of sferics and their intensity are measured as a function of time.

Major Functional Elements:

a. Antenna - A whip antenna is used.

b. Reciever - A broadband receiver is used.

c. Electronic data processing and storage - The data are sorted and stored for transmission at regular intervals.

Time of Operation: The sferics detector begins data collecting as soon as the Lander has landed (and oriented in the case of Mars).

Location: The sferics detector is located within the Lander or Orbiter and the whip antenna extends into space.

Dimensions and Power:

Weight: 3 lbs
Size: 4 in x 5 in x 6 in + whip antenna
Power: 2 watts

Relationship to Other Experiments: The data from the sferics detector will be correlated with the data from the anemometer, precipitation gauge, and TV pictures of sky and landscape.

Instrument No. (I-84)

DUAL LIGHT LEVEL INDICATOR

Objective: To measure the integrated visible solar flux incident upon the surface of a planet. To measure the light available for television viewing.

Principle of Operation: A pair of photoconductive cells, sensitive in the visible region, are used in series with a fixed resistor. A regulated voltage is applied to the circuit and the output is taken across the fixed resistor to produce an output of 0 - +5 vdc. One cell measures the general incident solar light intensity, and the other cell views the same scene as does the TV camera.

Parameter Measured: The radiation incident upon the detector is measured while the Lander is on the surface of the planet.

Dynamic Range Ratio: The dynamic range ratio will be 10^4.

Response: The rise time to a step function input will be less than 10 seconds.

Major Functional Elements:

a. A pair of photoconductive cells, sensitive in the visible wavelengths.

b. A thermistor to measure the temperature of the photocells. (Temperature correction may be needed for the photocell reading.)

Time of Operation: The dual light level indicator will operate during daylight in general, although it can be turned on at any time.

Location: The indicator is located on the TV camera in such a manner that one cell views the same scene as does the TV camera. The other cell views the sky hemisphere when the camera is viewing parallel to the planetary surface.

Dimensions and Power:

Weight: 0.3 lbs
Size: 2 in x 2 in x 2 in
Power: 0.1 watt

Relationship to Other Experiments: The cell viewing the sky hemisphere would be able to indicate the presence of overhead clouds. This indication can be used to turn on the TV to take a picture since the TV will not be on at all times. Secondly, it will indicate the duration of cloudiness and these data will be correlated with the measurement of quantity and type (liquid or particulate) of precipitation as measured by the precipitation gauge.

The second cell will indicate the presence or absence of sufficient light to take TV pictures.

Instrument No. (I-85)

RADIO PROPAGATION EXPERIMENTS (BISTATIC RADAR)

Objectives:

a. Determination of integrated electron density along direct and reflected paths.

b. Height and density of planetary ionospheric maxima.

c. Reflectivity and roughness parameters for reflection points on planetary surface.

d. Determination of dielectric constant.

e. Range and range-rate data to help determine planetary mass (especially Venus).

f. Ionospheric and atmospheric density, profile, and structure.

g. Value and change of integrated electron density during periods of solar activity.

h. Study feasibility of using radio propagation technique on Venus Orbiter to determine direct and reflected path measurements of density, structure, and dynamics of solar corona when Venus is in or near opposition. Use of relatively low frequencies probably of critical importance.

i. When not used in bi-static mode, radar may be used for other purposes (space probe tracking, telemetry, echoes from planet, etc.).

Principle of Operation: A powerful radar transmitter on Earth directs its signals towards a planet. Two signals are received on a space probe when near the planet: a direct ray and a ray reflected from the planetary atmosphere and/or surface. The reflected ray will not reach the receiver on the probe as rapidly as the direct ray. The delay times will be telemetered back to Earth. These data and the relative Earth-planet-probe positions will then be analyzed.

Parameters Measured: Phase and group velocities, polarization, and amplitudes will be measured for each of several radio frequencies between 20 and 2000 mc. 50 and 400 mc bands are currently planned for a PIONEER experiment. The addition of a few other appropriate frequencies will broaden the information which can be analyzed.

Major Functional Elements:

a. Large, steerable radar transmitters and receivers appropriately positioned on the Earth. Power: approximately 1 Mw.

b. All transmitters and receivers to analyze polarizations and amplitudes, as well as phase and group velocities.

c. Receiver and antenna on space probe.

Time of Operation:

a. On command during cruise.

b. Continually during orbit.

Location: Antenna on magnetometer boom.

Dimension and Power:

Weight: 15 lbs (including antenna)
Size: 4 in. x 4 in. x 12 in.
Power: 2 watts

Discussion: The radio propagation experiment provides information about many physical phenomena. When used to study electron densities in a planetary atmosphere, it is noted that integrated values or changes in values are obtained. Since this is the case, it is desirable to integrate the results of this experiment with others (e. g., atmospheric temperature, pressure, density and composition, intensity and polarization of skylight, bottomside sounder, and ultraviolet and infrared spectral and radiometric studies) in order to secure optimum utilization of the results. The electron density of interplanetary space can be nulled out when observing a planetary atmosphere because it will have essentially the same effect on both the direct and reflected rays. However, this technique also provides a method of determining integrated electron densities and changes in density of interplanetary space.

Instrument No. (I-87)

BOTTOMSIDE IONOSPHERIC SOUNDER

Objective: To study the nature of the Martian ionosphere.

Principle of Operation: The degree to which an electromagnetic wave can penetrate a region containing free charge is determined by the charge density and frequency of the wave. When a critical charge density is reached, the wave is reflected. By the proper selection of several frequencies, bursts of radio waves may be used to study the ionospheric charge density by observing the time required for the echo to be detected.

Parameters Measured: The sounder will measure the charge density distribution in the lower Martian ionosphere as a function of altitude about the Martian surface.

Major Functional Elements:

a. Radio frequency Generator and Transmitter - The generator and transmitter provide the radio wave pulse at the required frequency and power level.

b. Antenna - Two dipole antennas are used to transmit the outgoing pulse and receive the returning echo. Two antennas of different lengths are required to cover an adequately large range of frequency without excessive power loss.

c. Echo detector - The detector measures the time it takes for the echo to be received after the initial pulse has been transmitted. A high frequency clock is provided in the detector to supply the time base.

Time of Operation: Continually at periodic intervals (two times per hour) for several days per month.

Location: Electronic equipment may be placed anywhere within the landing vehicle. Antennas must be extended parallel to ground surface.

Dimensions and Power:

Weight: 50 lbs total, including antennas
Size: electronics - 12 in. x 12 in. x 12 in.
antennas - 400 ft and 200 ft dipoles
antennas erectors (4) -- 5 in. dia. x 12 in. each
Power: 25 watts, intermittent

Discussion: It has been assumed that the Martian surface is a sufficiently poor conductor that the dipole antennas may be wires dispersed by a mortar and left lying on the ground.

Instrument No. (I-90)

ACTIVE SEISMIC EXPLORER

Objective: To determine local layering, existence of isostatic compensation of topographic features, possible locations of useful minerals, and variation in crustal thickness.

Principle of Operation: Geophones are deployed and explosive charges are subsequently deployed and set off. Seismic waves are reflected from density discontinuities and their propagation velocity undergoes dispersion which is characteristic of the density distribution with depth.

Parameters Measured: The motion of the planetary surface is measured as a function of time after the explosion of a charge, of charge size, and of charge location relative to the geophones.

Major Functional Elements:

a. Sensors — Sensors will be conventional moving coil geophones laid out in an array.

b. Geophone Distributor — A mortar-like device to throw out a line with the attached geophones and to form a linear array of geophones.
c. Charge Distributor — A mortar-like device to toss the charges to the desired location.
d. Electronics Package — This contains an amplifier, time interval measuring circuits and data storage.

Time of Operation: The active seismic explorer will be activated about four days after the Lander has been oriented. This time may be changed by Earth command.

Location: The main body of the instrument will be within the Lander. Provision will be made for the deployment of geophones and explosive charges.

Dimensions and Power:

Weight: 90 lbs
Size: 4 geophones and deployment, 4 in. dia. x 6 in. long
4 charges and deployment, 6 in. dia. x 3 in. long
Electronics, 8 in. x 10 in. x 12 in.
Power: 5 watts

Special Limitations: In order to locate the relative positions of charge explosion and geophones, the TV will be used to view the scene. The charges will be fused, so that there will be an interval of time between the deployment of the charge and the explosion. The TV will locate the distinctively-colored geophones and charge.

Relationship to Other Experiments: The TV will view the dirt geyser produced by the charge. Knowing the charge characteristics, and the dirt thrown up, some knowledge may be had regarding surface trafficability. It may not be exceptionally useful data, but it is essentially free.

Instrument No. (I-91)

SEISMOMETER, 3-AXIS

Objective: To obtain the microseismic activity of a planet; this activity being an indication of the thermal state of the planet and its tectonic activity.

Principle of Operation: For all details, refer to the one-axis seismometer (I-21). Basic design principles and all but the external characteristics of the instrument remain the same.

Dimensions and Power:

Weight: 34 lbs
Size: 10 in. dia. x 15 in. long
Power: 4 watts

Instrument No. (I-93)

AIRGLOW ANALYZER, SPECTROMETER

Objective: To obtain the spectral distribution of the radiation emitted by the planetary atmosphere.

Principle of Operation: A spectrometer will be used covering the wavelength range from 3900 Å to 7800 Å and having a 10 Å resolution. This spectrometer will view the planetary atmosphere.

Dynamic Range Ratio: The dynamic range ratio of the intensity will be 10^3.

Response: The spectrum will be swept in less than 10 seconds.

Location: The spectrometer will be located on the planet horizontal package of the Orbiter.

Time of Operation: The spectrometer will be used after the Orbiter is in orbit.

Dimensions and Power:

Weight: 22 lbs
Size: 12 in. dia. x 15 in. long
Power: 12 watts

Relationship to Other Experiments: The data from this experiment will be correlated with those data taken during entry from the UV spectrometer mounted on the Lander.

Instrument No. (I-94)

SOUNDING ROCKET

Objective: To investigate the upper atmosphere of Mars.

Principle of Operation: A solid fuel sounding rocket will carry five pounds of payload to an altitude of 200,000 feet. The payload will make measurements continuously during the flight until it is destroyed by impact on the surface.

Major Functional Elements:

a. Motor — The motor will provide approximately 1000 lb-sec total impulse to propel the rocket to the desired height.
b. Transmitter — The transmitter will telemeter the experimental information to the Lander in an extremely simple form (e.g., pulse width and repetition rate of carrier frequency can carry two channels of information).
c. Experiment Package — Atmospheric temperature and pressure will be measured.
d. Battery — A small battery will be used to supply power to the experiment package and transmitter.
e. Firing Tube — The firing tube will guide the rocket during launch.

Time of Operation: After landing, after TV pictures have been obtained, and after biological/petrographic experiments have been completed.

Location: Location is not critical but rocket should be fired upward within several degrees of local vertical.

Dimensions and Power:

a.	Weight:	Rocket	11 lb (not incl. payload)
		Payload	5 lb (exp., transmitter and battery)
		Firing Tube	3 lb
		Total	19 lb.

	Size:	3.5 in. dia. x 30 in. long plus 4 fins at top of firing tube, each 9 in. x 9 in. (dimensions include firing tube)
	Power:	25 watts (momentarily for ignition)
b.	Weight:	Lander receiver 15 lbs (incl. antenna)
	Size:	3 in. x 8 in. x 8 in. (not incl. antenna)
	Power:	3 watts

Relationship to Other Data: The data taken with the sounding rockets will be correlated with those data taken by the Lander during entry and descent.

Instrument No. (I-98)

SURFACE SAMPLER (PNEUMATIC)

Objective: To obtain a surface dust sample for biological analysis.

Principle of Operation: The blowing of "air" upon the Martian surface accompanied by simultaneous aspiration nearby permits the gathering of aerosolized dust particles from the surface. Collection may be accomplished by filtration or impaction.

Parameter Measured: The sampler performs no measurements. It gathers dust samples which may then be studied by other devices.

Major Functional Elements:

a. Blower — A vane-axial blower provides the pressure and suction to operate the collection system.
b. Transport Tube — The transport tube consists of two concentric collapsible tubes. The inflowing air and dust travel in the inner tube to the collector and blower; the effluent air travels in the annular space between the inner and outer tubes to the aerosolizing jets. A helical coil spring, also in the annular region, is used to extend and strengthen the transport tube assembly. The tube can be stretched to a maximum length of 10 feet.
c. Collector — All the incoming air passes through the collector. Dust is removed from the air by either an impactor of filter.
d. Aerosolizer — The aerosolizer assembly consists of the aerosolizing jets, the sample intake ports, a set of pneumatic tires (inflated by effluent from blower) and a small electric motor. The last two items provide some mobility to the aerosolizer.

Time of Operation: After landing, for several minutes for each sample acquired.

Location: The sampler must be located such that it may drop from the vehicle onto the Martian surface.

Dimensions and Power:

Weight: 2 lbs
Size: 3 in. dia. x 10 in.
Power: 5 watts

Instrument No. (I-99)

AEROSOL PROFILE METER

Objective: To obtain the vertical distribution density of aerosols in the atmosphere of Mars.

Principle of Operation: While the Lander is descending, two types of filters will be exposed to the atmosphere which will be passing by at a speed of the order of 1000 ft. sec^{-1}. One type of filter will trap and hold solid particles and the other type of filter will detect moisture by an irreversible dye color change. At 10 second intervals, the exposed filters will be retracted and stored and new filters will be exposed. After the Lander has landed and has oriented itself, the stored filters will be examined by the microscope and TV pictures telemetered to Earth via the communication system.

Parameters Measured:

a. Solid particulate material
b. Water particles

Major Functional Elements:

a. Filter Movement — A mechanism exposes and retracts filters at 10-second intervals. Later, on the surface the filters are put under the microscope and then back into storage or discard.
b. Filter Storage — The filters will be stored in sequence so that the data contained in them may be correlated with altitude.

Time of Operation: The filters will be exposed during the Lander's descent and read out after the Lander has reached the planet's surface.

Location: The aerosol profile meter is located on the periphery of the aft cover.

Dimensions and Power:

Weight: 3 lbs
Size: 1/2 in. x 1 in. x 6 in. probe + 6 in. x 6 in. x 6 in. storage and electronics
Power: 2 watts

Relationship to Other Experiments: The data from this experiment will be correlated with data from the radar altimeter and the other atmospheric measurements.

Instrument No. (I-100)

LASER-SURFACE SPECTROMETER ASSEMBLY

Objective: To obtain knowledge of the composition of planetary crust material.

Principle of Operation: A laser beam is focused upon a portion of the planetary crust material. The energy vaporizes the material and thus causes it to be incandescent. The emission spectrum of this incandescent vapor is obtained with a spectrometer. Several cycles may be required, depending upon the crust material.

Parameter Measured: The spectrum of the incandescent planetary crust material is obtained.

Major Functional Elements:

a. Laser — A 2-3 joule ruby laser is used.
b. Capacitor Bank — A capacitor bank to store 1000 joules is used.
c. Spectroscope — A spectroscope obtains the spectrum of the vaporized material.

Time of Operation: The laser-surface spectrometer assembly will be turned on 36 hours after the Lander has been oriented. The time may be changed by Earth command.

Location: The laser-surface spectrometer assembly will be located within the Lander, and a sample will be brought to it.

Dimensions and Power:

Weight: 50 lbs
Size: 20 in. x 18 in. x 22 in.
Power: 2 watts

Relationship to Other Experiments: The data will be correlated with those data from the other soil analysis experiments.

Instrument No. (I-101)

LASER ATMOSPHERIC BACKSCATTER PROBE

Objective: To obtain data on the dust contents of the planetary atmosphere: the amount of dust as a function of altitude.

Principle of Operation: An intense light beam of short duration obtained from a Q-switched laser is directed upward from the planetary surface. The light which is backscattered by atmospheric-borne particles is detected and measured as a function of time. From these data, the amount and altitude of dust may be inferred.

Parameters Measured: The time dependence and amplitude of backscattered light will be measured.

Range: Backscatter from altitudes to 15-20 miles may be obtained, depending upon the density vs. altitude distribution of the dust.

Major Functional Elements:

a. A Q-switched 2-3 joule laser will be used.
b. Capacitor Bank — A 1000 joule capacitor bank will be used.
c. Light Detector — A multiplier phototube light detector will be used to obtain the time dependence and amplitude of the backscattered light.

N. B., This experiment will use the same capacitor bank, power supply, and laser as is used in the laser-surface spectrograph assembly in order to avoid duplication of weight.

Time of Operation: The laser-atmospheric backscatter probe will be used at night. It will be operated once every hour for eight hours each night for 30 days. The time of operation may be changed by Earth command.

Location: The bulk of the apparatus will be within the Lander. The laser will be mounted on top of the Lander and will be beamed in the vertical direction. The light receptor will be mounted on top of the Lander to view the backscattered light from the vertical.

Dimensions and Power:

Weight: 20 lbs (Not including the capacitor bank, laser and power supply).
Size: 10 in. x 10 in. x 8 in. (excluding the laser, power supply, and capacitor bank).
Power: 15 watts while in operation; 2 watts for 15 minutes and 13 watts additional for 3 minutes.

Relationship to Other Experiments: The data from this experiment will be correlated with that from the various experiments sensing light intensity levels during the daytime.

APPENDIX A. EXPERIMENTS

During the course of the Voyager study a number of experiments have been specified in relatively complete discussions. While these discussions are not entirely necessary for including the instruments in given payloads, and indeed some of the experiments have not even been included in the missions suggested, they are useful background material for general measurement planning. It is entirely possible that some of them will be included in the actual measurement systems.

1. COSMOLOGICAL EXPERIMENTS

This discussion of cosmological measurements was written entirely by Dr. Ralph Alpher, General Electric Research Laboratory, Schenectady, N. Y., who was one of the scientific consultants in the Voyager study. Dr. Alpher's extensive background in cosmological problems assures the validity of the ideas expressed.

At present the basic cosmological experiments appear to be those which would provide information to help in distinguishing between the various currently proposed cosmological models, viz:

The homogeneous isotropic steady-state universe;
The homogeneous isotropic evolutionary universe;
The homogeneous isotropic universe in which the "constants of nature," and in particular the gravitational "constant" may vary in space, time, or both;
The anisotropic universe, steady-state or evolutionary.

It is clear that almost any set of observations one makes of a general astrophysical nature will ultimately contribute to resolution of cosmological questions. For example, any improved picture of stellar structure and evolution should improve one's use of variable stars and "brightest" stars in establishing extragalactic distance scales. This in turn will improve one's estimates of the distribution of matter in the universe and help distinguish between cosmologies. Among such stellar studies are ultraviolet and X-ray spectroscopy of stars from outside the earth's atmosphere.

Experiments which may be of somewhat more direct cosmological interest and might be performed from a non-recoverable space probe are listed below. Some of these are described in reviews by Goldberg and Dyer (1960), and by Dicke (1960):

(a) Sky Reconnaissance at Wavelengths Inaccessible From the Ground

Here one is particularly interested in determining at various wavelengths the separate contribution of unresolved extragalactic radiation sources to the brightness of the sky. Quantitative generalized sky brightness due to all extragalactic sources would permit a quantitative discussion of Olber's paradox. This paradox has to do with why the sky we see is not as bright as the surfaces of all the radiating bodies in the universe, and resolution of the paradox provides the basic assumptions for cosmological theories. In a more practical vein, the spectroscopic study of galaxies at recession velocities of ~ 0.2c is greatly limited by the confusion of the unresolved background radiation. And it is just at ~ 0.2c that deviations from linearity in the expansion become non-trivial (Baum 1957). It may be noted that McVittie (1962) has shown that the difference in expected unresolved radiation background in the steady state and evolutionary cosmologies is disappointingly small. Wavelength regions of interest and inaccessible from earth would be $\lambda < 0.3\mu$, $24\mu < \lambda < 3$mm, and 15m $< \lambda$. The infrared regions 0.8 to 24μ would be of great interest even though there are some atmospheric windows for the reason that there may be a high background from distant galaxies with high recession velocities and hence large red shifts. It would appear that broad band photometric studies would be required since one would depend on telemetry. Clearly one would have to know vehicle orientation rather well in order to rule out sources of radiation in the solar system. It might be possible to utilize the radiation detectors already planned for planetary studies to do these background radiation measurements. If there is a large enough range of sensitivity, one could obtain at least upper limits and perhaps better. It is not clear how one makes any positive statements as to the intensity of the unresolved background radiation other than the obvious one that it must be weak. There is the inference from Baum's paper (1957) that optical spectroscopy of galaxies of 17th or 18th magnitude and beyond is seriously hampered by the unresolved background. (The visual illuminance from a zero-magnitude star outside the earth's atmosphere is estimated to be about $2.65 . 10^{-10}$ lumens/cm^2, so an 18th magnitude background would be $(2.512)^{-18} \times 2.65 . 10^{-10}$ lumens/cm^2.) Unfortunately, coupled with this inherently low level of measurement is the seeming requirement that the aperture of the observing devices be quite small in order to facilitate separation of resolved and unresolved radiation sources (McVittie 1959). (See also Davidson (1962).)

(b) Relativity and Gravitation

Attractive as it may be to hope for observational tests of those cosmological theories in which the gravitational constant and/or other fundamental constants are not in fact constant, no direct experiments suggest themselves for Voyager. (There may be some feasible experiments for a terrestrial or solar satellite.) There are, however, several types of experiments having to do with time dilatation in special relativity and the gravitational "red" shift in general relativity (actually one checks at most the principle of equivalence) which might be done with the same equipment. In part, the possibility depends on the existence of commercially available rubidium vapor frequency standards (Varian V-4700A) whose stability has been characterized in terms of the standard deviation of the frequency difference of two standards, viz., 5×10^{-11} over a one-year period, and 1×10^{-11} for one-second averaging time. This device appears to meet the requirements for an "ideal standard clock." (Møller, 1955). The experiment also would depend on a knowledge along the trajectory of the velocity of the spacecraft and of the gravitational potential in which the vehicle finds itself.

Thus, if χ represents the local gravitational potential as a function of coordinates and time, with χ_o the value at the earth's surface, the "ideal clocks" as free running atomic oscillators on the spacecraft and on earth should show a relative frequency shift:

$$\frac{\Delta\nu}{\nu} = \sqrt{1 + \frac{2\chi}{c^2}} - \sqrt{1 + \frac{2\chi_o}{c^2}}$$

$$\cong \frac{-(\chi_o - \chi)}{c^2}, \qquad \chi/c^2,\ \chi_o/c^2 << 1,$$

and one would hope to verify the gravitational shift by comparisons over short time durations (during which χ is sensibly constant) of the frequency of the spacecraft oscillator and the earth-bound oscillator. It would be necessary to correct the telemetered signal for comparison for several competitive effects. Note that when χ is sufficiently small,

$$\frac{\Delta\nu}{\nu} \cong -7 \times 10^{-10}$$

so that the rubidium vapor standard appears to be a feasible device.

A comparison of clock rates, and hence indirectly of time dilatation, could in principle also be made in a short duration measurement. The rate difference between the earth-borne and space-borne clocks is given by:

$$\frac{\Delta\omega}{\omega} = \sqrt{1 + \frac{2\chi}{c^2} - \frac{v^2}{c^2}} - \sqrt{1 + \frac{2\chi_o}{c^2}}$$

where v is the velocity of the spacecraft relative to the earth-borne clock. Note that a velocity of 11 km/sec should yield a frequency shift of 7×10^{-10}. Higher precision and a more meaningful experiment would result from an integrated measurement. Thus one would measure the time duration in the two systems (at a given time t) in terms of the number of elapsed cycles of the two clocks and compare them as

$$N_V - N_e \cong \int_0^t \left(\frac{\chi - \chi_o}{c^2} - \frac{1}{2}\frac{v^2}{c^2} \right) dt.$$

As Dicke has mentioned, should sufficient precision be achieved in this experiment, one might then interpret any deviation from expectation in terms of a departure from constancy in some of the atomic constants. However, except for this very remote hope, it is doubtful in the writer's opinion that one could demonstrate with "clock experiments" on Voyager the various relativity-associated effects as well as, let alone better than, has already been done with Mössbauer-effect experiments. (Frauenfelder, 1962)

(b) Cosmic Gamma Ray Flux

The one experiment of rather direct cosmological interest which can be performed from Voyager vehicles en route to Venus or Mars is a measurement of the cosmic gamma ray flux. This is in a sense a special case of the "Olbers' background radiation" measurement. Cosmic gamma ray measurements have already been made from balloons, (Cline, 1961; Duthie, et al, 1963), rockets (Giacconi, et al, 1962), satellites, (Kraushaar and Clark, 1962) and from a space probe, (Arnold, et al, 1962). There have been a variety of interpretations of the cosmic gamma ray flux, of which the most interesting appears to be that of Felten and Morrison (1963) — who identify the source of the gamma rays as an inverse Compton effect in the collision of starlight photons and the fast electrons whose presence one deduces from non-thermal radio emission. Whether the Felten-Morrison mechanism is dominant or whether there is some other phenomenon giving rise to cosmic gamma-rays, there seems little doubt that the gamma ray data will provide clues to the amount and distribution of matter in extragalactic space. Since a more reliable figure for the mean density of matter in the universe is a basic need in cosmological theories, there can be no doubt of the utility of gamma ray measurements.

Comments on Gamma-Radiation Studies

Measurement of the gamma-ray component of the primary cosmic radiation has been an area of interest in cosmic ray physics for many years. It has been a difficult thing to accomplish because gamma rays also are produced as secondaries in interactions of the particle component of the cosmic radiation with the material in the earth's atmosphere. Moreover, not only does this secondary production mask primary gamma radiation traversing the atmosphere, but because of backscatter one sees the secondaries in measurements close to but above the atmosphere. This secondary albedo is, therefore, still a problem for gamma-ray measurements aboard balloons, rockets, or earth satellites and is an important reason for measurements aboard vehicles which get well away from the earth's atmosphere. While Voyager is not intended primarily as a deep space probe, it nevertheless will spend a considerable time en route to Venus or Mars, during which time it can be a most effective vehicle for this purpose.

Before a brief description is provided of the experiments that have already been done, it may be of interest to reiterate some of the possible sources of "primary" gamma radiation (Rossi, 1961).

(a) Collisions of Particulate Cosmic Radiation with Matter Other Than That in Our Atmosphere

In high energy collisions there are produced π^o-mesons among whose decay products are gamma rays. Possible locations for such collisions include interstellar matter in our galaxy, and matter in nearby galaxies such as in the Magellanic Cloud, material in intergalactic space, and of course the solar atmosphere. Moreover, in those strong radio sources such as the Crab Nebula, Cassiopiea A and Cygnus A the same mechanism that accelerates electrons and leads to syncrotron emission may also accelerate protons, which in collisions would lead ultimately to gamma rays. These radio sources would be point sources of gamma radiation for any feasible spacecraft detector.

(b) Bremsstrahlung and Synchrotron Radiation

Provided there is somewhere a source of fast electrons (galactic halos or hot interstellar plasma) and a magnetic field (prevalent in the galaxy and near many stars), one could obtain a flux of gamma radiation via bremsstrahlung and/or synchrotron radiation, respectively. Hoyle (1963) has presented some reasonably convincing arguments against such possibilities as the source of ~ 5 kev radiation (Giacconi, et al, 1962) and suggests instead that one consider the steady-state model of the universe, with intergalactic space populated by "created" neutrons and the products of neutron decay. He finds the resultant hot plasma capable of providing the observed ~ 5 kev radiation.

(c) "Background Radiation"

The possibility presents itself that the observed gamma ray flux is a portion of the spectrum of the integrated output of all radiation sources in the universe. McVittie and Wyatt (1959) and McVittie (1962) have discussed this problem in terms of a comparison of expectations for the two principal cosmological theories — the evolutionary and steady-state theories. At this stage a number of assumptions are required to compare these theories with observed gamma ray fluxes — in particular some rather violent assumptions are needed on the source function — the radiation properties of the average galaxy. McVittie finds either theory does about as well with the ~ 100 Mev flux measured by Kraushaar and Clark (1962) but neither theory does very well in terms of reasonable values of resulting mean universal densities.

(d) Nucleon-Antinucleon Annihilation

Such annihilation produces gamma rays through the intermediate step of π^o-mesons. Nucleon-antinucleon pairs might be expected to be uniformly and widely distributed in a steady state cosmology as the form in which matter is created, or they might be the residue of primary formation processes in an evolutionary universe. There is reason to believe that antimatter cannot exceed one part in 10^7 of ordinary matter and probably the limit is considerably smaller. (Alpher and Herman, 1959; Grigorov, et al, 1962; Kevane, 1961).

(e) Inverse Compton Effect Between Stellar Photons and Fast Electrons (Felten and Morrison, 1963)

Fast electrons undoubtedly exist in galactic halos where they are the source of the non-thermal syncrotron emission observed by radio telescopes, and such fast electrons may well be present in space albeit previously unobserved. This inverse Compton effect is discussed further below.

(f) Residual High Energy Photons from an early state of high radiation density in an evolutionary universe.

Estimates can and have been made of possible contributions from most of this variety of possible sources. Some can be ruled out as below detectability, others would be distinguishable in terms of isotropy or anisotropy in measurements that are made. Measurements thus far seem to indicate a finite isotropic primary gamma ray component, with a specific energy spectrum (Felten & Morrison, 1963) although clearly many more measurements need to be made (Bhabha, 1963). Moreover, the existence of these data make it now less necessary to estimate flux sources since the required counting rate levels are probably well enough known for the design of future experiments.

As is perhaps clear from the discussions above, the writer feels the analysis of existing gamma ray flux measurements by Felten and Morrison is the most attractive yet. They consider the gamma radiation as being the recoil photons resulting from collisions of starlight photons with fast (relativistic) electrons. They find that considering only the electrons in the galactic halo yields a flux energy spectrum of apparently the observed slope but with a factor of 300 too low an amplitude. To adequately represent the data would require either an absurdly high electron density in the galactic halo or an intergalactic content of fast electrons (perhaps leakage from galactic halos, but a highly interesting possibility in any event) of the order of a percent of the halo electron density. Observation of such electrons would have been difficult before now and in fact observation via the inverse Compton effect is probably just the way of looking for such electrons. In any event, the Felten-Morrison theory is surely speculative and many more gamma-ray measurements are needed.

Reference Material on Gamma Ray Experiments

With regard to planning gamma ray measurements aboard Voyager spacecraft, the following reference material contains descriptions of the actual experiments performed aboard other vehicles to date at the indicated energies. It is likely that these experiments could be adapted to Voyager.

$\gtrsim$ 50 Mev	Rossi (1961); Explorer XI Kraushaar & Clark (1962); Explorer XI Duthie, Hafner, Kaplon and Fazio (1963); balloon Fazio and Hafner (1961); balloon T. L. Cline (1962); balloon
1 Mev	Arnold, Metzger, Anderson and Van Dilla (1962); Ranger
5 Mev	Giacconi, Gursky, Paolini and Rossi (1962); Aerobee rocket

BIBLIOGRAPHY

J. R. Arnold, A. E. Metzger, E. C. Anderson and M. A. Van Dilla, "Gamma Rays in Space, Ranger 3," J. Geophys. Res. 67, 4878 (1962).

W. A. Baum, "Photoelectric Determination of Redshifts Beyond 0.2c," As. J. 62, 6 (1957).

H. J. Bhabha (private communication)
That much remains to be done in evident from preliminary results obtained in India at the Tata Institute. In a study of extensive air showers due to primary particles with energies above 10^{16} ev, there has been found a N-S asymmetry in the orientation of shower occurrence origins which has as yet not been correlated with any local or large-scale features of the environment. Better gamma ray measurements may reveal anistropy whence a correlation of counting rate with location and orientation of the experiment is most desirable.

T. L. Cline, "Search for High-Energy Cosmic Gamma Rays," Phys. Rev. Letters 1, 109 (1961).

W. Davidson, "Radio Background as a Criterion for Cosmological Models," Nature 196, 155 (1962).

R. H. Dicke, "The Nature of Gravitation," Chapter II of Science in Space - A Report by the Space Science Board, published by the National Academy of Sciences, National Research Council, Washington, D.C., 1960.

J. G. Duthie, E. M. Hafner, M. F. Kaplon and G. G. Fazio, "Gamma Rays at High Altitudes," Phys. Rev. Letters 10, 364 (1963).

G. G. Fazio and E. M. Hafner, "Directional High Energy Gamma-Ray Counter," Rev. Sci. Inst. 32, 697 (1961).

J. E. Felten and P. Morrison, "Recoil Photons from Scattering of Starlight by Relativistic Electrons," Phys. Rev. Letters 10, 453 (1963).

Hans Frauenfelder, The Mössbauer Effect, (W. A. Benjamin, Inc., New York, 1962). Chap. 5.

R. Giacconi, H. Gursky, F. R. Paolini, and B. B. Rossi, "Evidence forX-rays from Sources Outside the Solar System," Phys. Rev. Letters 9, 439 (1962).

Leo Goldberg and Edward R. Dyer, Jr., "Galactic and Extragalactic Astronomy," Chap. VIII of Science in Space - A Report by the Space Science Board, published by the National Academy of Sciences, National Research Council, Washington, D.C., 1960.

N. I. Grigorov, D. A. Zhuravlev, M. A., Kondrat'eva, I. D. Rapoport and I. A. Savenko, "Searches for Antimatter in Cosmic Rays and Outer Space," Planet Space Sci. (GB) 9, 875 (1962).

F. Hoyle, "X-rays from Outside the Solar System," Ap. J. 137, 993 (1963).

C. J. Kevane, "On Antimatter and Cosmology," Science 133, 580 (1961).

W. L. Kraushaar and G. W. Clark, "Gamma Ray Astronomy," Scientific American, p. 52, May 1962.

W. L. Kraushaar and G. W. Clark, "Search for Primary Cosmic Gamma Rays with the Satellite Explorer XI," Phys. Rev. Letter 8, 106 (1962).

G. C. McVittie, "Photon Density and the Gamma-ray Flux at a Point in an Expanding Universe," Phys. Rev. 128, 2871 (1962).

G. C. McVittie and S. P. Wyatt, "The Background Radiation in a Miine Universe," Ap. J. 130, 1 (1959).

A. E. Metzger, M. A. Van Dilla, E. C. Anderson and J. R. Arnold, "Nuclear Missions in Progress. Ranger Gamma-ray Spectromener," Nucleonics 20, 64 (1962).

C. Møller, "Old Probems in the General Theory of Relativity Viewed from a New Angle," Det. Kong. Dansk. Vid. Sels. Mat.-Fys. Medd. 30 (10) 1955.

B. Rossi, "Some Plans for Experiments in Space," in Space Astrophysics, W. Liller (Ed.) (McGraw-Hill Book Co., New York, 1961) p. 101.

2. PLANETO-PHYSICAL EXPERIMENTS

a. Analysis of Elemental Composition of Planetary Crust by Neutron Activation

The purpose of this experiment is to supplement compositional analysis of planetary crusts resulting from the α-particle scattering experiment. Compared to the latter, the present experiment extends the range of elements to higher mass numbers ($\approx$ 60). Light elements (below oxygen) cannot be determined by this method because the resulting nuclides have either short half-lives or emit γ rays too soft for practical detection.

Experimental Method

The neutron activation method is based on the fact that bombardment of a sample by neutrons results in the production of radioactive nuclides in the target material. The decay of these nuclides is generally accompanied by the emission of γ and β rays having energy spectra characteristic of the nuclides in question. Measurement of this spectrum permits, at least in principle, the identification both qualitative and quantitative of the elements comprising the original target material.

Remarks on Practical Application of the Method

The basic principle of the neutron activation method is deceptively simple. However, in practice a large number of factors conspire to make the measurement difficult, especially when the target is chemically complex. A few of the factors which have significant bearing on the sensitivity and precision of the method are: irradiation time, decay time, competing reactions, half-life of the product, energy of γ or β rays used for identification, interfering γ rays from products of no interest, external scattering into the detector, geometry (detector-target), self-shielding in sample, inhomogeneity in flux. For instance, irradiation time for optimum activity can range from a few seconds to several hours with a source of moderate neutron flux. A similar condition exists with the decay time. It should also be pointed out that with a scintillation detector normally used, a monoenergetic flux of γ rays produces a continuous pulse height spectrum characteristic of the γ ray in question. For a polyenergetic γ ray flux which will normally exist in a space application, the pulse height spectrum is a summation of spectra of monoenergetic components. The more elements one seeks, the more complex is the resulting spectrum making the interpretation a challenging task.

In view of the above, detailed characteristics of the instrument used in such an analysis depend strongly on the characteristics of the elements being sought.

Therefore, the following discussion must be considered merely as a crude guide to the instrument configuration, realizing that specific details have no claim to practical reality.

Instrumentation

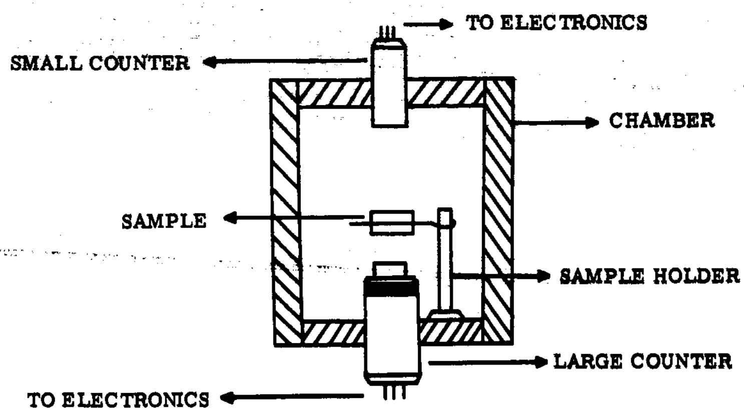
Equipment for the neutron activation analysis consists basically of three blocks. First of these is a neutron source needed for irradiation of targets. The second component consists of an arrangement of scintillation detectors measuring the γ-ray flux emitted by the product nuclides. Thirdly, the output of counters is processed by pulse height analyzers to determine the γ ray spectrum.

Source

In recent years a number of manufacturers introduced small sized neutron sources producing a flux of 10^7 - 10^{10} neutrons/sec in the 14 Mev energy range. A representative size of the tube containing such a source is 16" long and 2.5" in diameter. Accelerating system requires potentials in the 100 to 200 kv range. The power needed for operation is 25-30 watts. Sources can be operated continuously or they can be pulsed. Among commercial manufacturers active in this field, one can mention Phillips Research Laboratories (Holland), Schlumberger Well Surveying Corporation, and Kaman Aircraft Corporation.

Counter Arrangement

After irradiating the target for a suitable length of time, the target is transferred to a counting device. A possible arrangement is shown in the following sketch.



The detectors typically employed are NaI (T1) and anthracene crystals canned and mounted on phototubes. Commercial examples of such detectors are Tracerlab RLD-2X and RLD-2 scintillation counters. High voltage to counters can be derived from Tracerlab RLT-7 power supply. A prepared sample may be required.

Electronics

The output of the counters is coupled through cathode followers (e.g., White) to linear amplifiers (e.g., Nuclear Enterprise). The amplified signal is then processed by either multiple channel pulse height analyzers or by a sliding channel analyzer. The output of the analyzers are then telemetered to Earth. Information capacity should amount to several thousand bits per spectrum.

An attempt to adapt this technique to space work was carried out by Well Surveys, Inc. and is described in their reports to the Jet Propulsion Laboratory. The project has not gone much beyond the feasibility study, and no hardware development was undertaken.

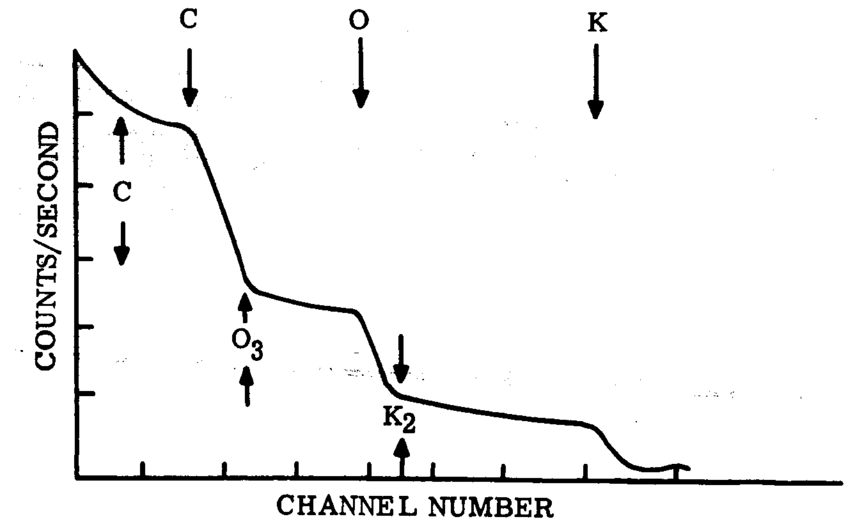
b. Chemical Composition of Crust by Scattering of α-Particles

Objectives:

The purpose of this experiment is to measure the chemical composition of the planetary crust material. The measurement is expected to identify all major elements with the exception of hydrogen. Individual elements can be identified through calcium. Elements of higher atomic mass can be identified only in groups.

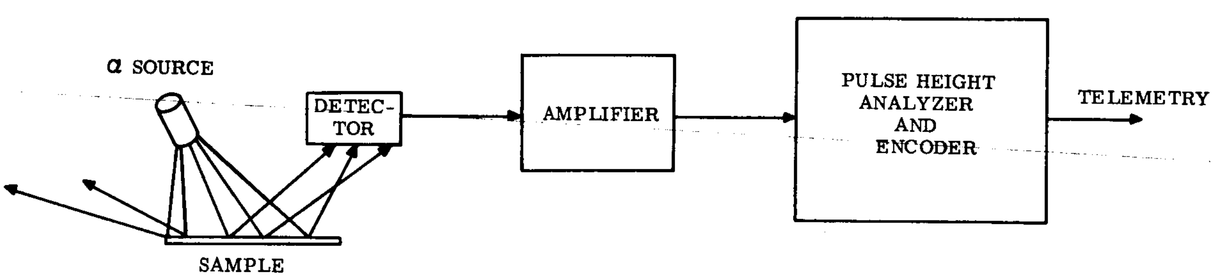
Basic Principles

The technique of compositional analysis of scattering of α particles is based on the fact that the energy spectrum of scattered particles depends on the mass numbers of nuclei within the target material and inversely on the atomic stopping powers for charged particles of the target material. For thick targets a continuous spectrum is obtained exhibiting sharp high energy cutoff edges whose position is unaffected by the chemical or physical state of the scatterer. For elements heavier than aluminum and low energy α particles, large angle scattering is primarily Rutherford's scattering. For lighter elements and higher energy α particles (α, p) reactions become important. The spectrum of the resulting protons contains information similar to that obtainable from α particle spectrum. The breakpoints in the spectrum identify the element and heights between plateaus are a measure of relative elemental abundances. A schematic spectrum for K_2CO_3 is shown in the following sketch.



Instrument

Schematic representation of a possible α-scattering instrument is shown below.



Typically, a sample is irradiated with 6 Mev α particles from a suitable source and the particles scattered at a large angle ($\approx 160^{\circ}$) are measured by solid-state surface barrier detectors. Similar detectors can be employed to detect protons. The resulting pulses are amplified and the conventional pulse height analysis is employed to determine the energy spectrum.

More detailed characteristics of a possible instrument designed to utilize both the proton and α-particle spectrum are shown in Table A-1.

Interpretation of data requires that a library of standard spectra for individual elements as well as simple compounds be available. These spectra are then matched to measured spectra by whatever means are either most convenient or readily available.

References

Turkevich, A., Science, 134, p. 672, September 8, 1961

TABLE A-1: SPECIFICATIONS FOR POSSIBLE INSTRUMENTATION FOR **ANALYSIS OF SURFACE MATERIAL BY** PARTICLE SCATTERING

Experiment, Priority	
Chemical Analysis By α-Particle Scattering	
Size	
Sensor head 4.25x4.5x2.1 inches Electronics 7x8x4.5	
Average Bit Rate	Bits/event
α-channel 2200 Proton channel 550	Total bits/sample depend on the length of time one wants to measure the sample
Weight	Power
Sensor head 1.3 lb Electronics 5.0 lb	1.5 watt (60 w-hrs)
Temperature Restrictions For	
Reliable Data	Permanent Damage
Too early to tell	Remains to be determined

Instrument	Special Requirements Such as Booms or Antennas	**No. of Directional Channels, Stabilized Vehicle**	**Reference to Instrument Design**
Consists of a sensor head containing 4 curium 242 sources and a detector for α particle scattering, and an α particle source, and four proton detectors. Two 200 channel pulse height analyzers are required.	For best performance the sensor head should be operated in vacuum. Thus, a vacuum chamber is required.	**Not applicable** **No. of Directional Channels, Spinning Vehicle** **Not applicable**	**Science, Vol. 134, 672, September 8, 1961** **University of Chicago work on contract to JPL**

Problem Areas	Hardware Development Status	**Orbit Preference**	**Comments**
The major problem area is the interpretation of complex spectra in terms of previously defined standards.	A bread-board model is under development by the University of Chicago for the Jet Propulsion Laboratory.	**Not applicable**	**Adjacent elements in the periodic table can be resolved to about mass number 40.** **Minimum detectable limit appears to be about 1% by atom.** **Carbon and oxygen exhibit enhanced scattering due to resonance effects.**

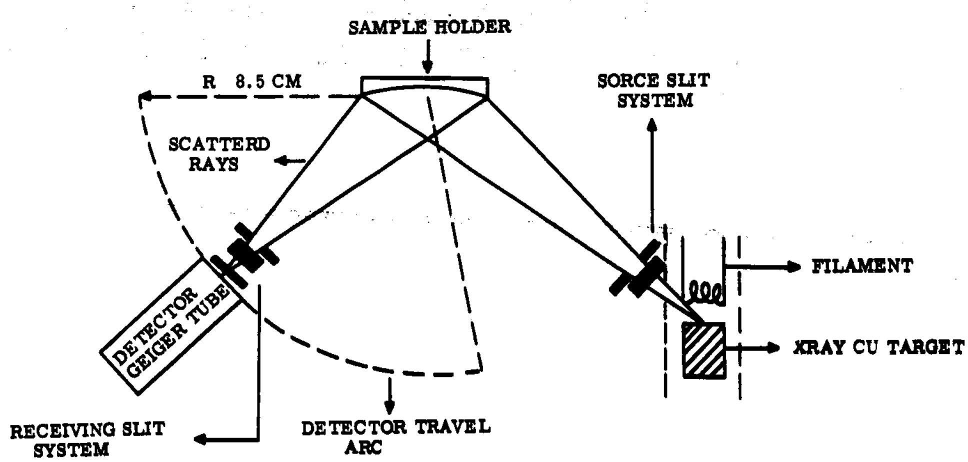
TABLE A-1 (CONT.)

Description of Transducer	Nature of Transducer Output (Analog or Digital)	Resolution Desired	Dynamic Range	Response Time (Fastest Expected)	Duration of Phenomena Being Measured
Solid state surface barrier or depletion layer detectors sensitive to α particles and protons. Proton detectors covered with a foil to stop scattered α particles.	Electrical pulses	Minimum detectable limit to be 1% by atom of the overall composition. At least 200 channels required in the pulse height analyzers to resolve elements up to Ca without ambiguity.	For α particle section 0-3000 cps per channel For Proton section 0-400 cps per channel	Equipment to handle 3,000 counts per sec without distortion, saturation, etc.	Not applicable. The total length of time per measurement is set by the precision with which one desires to measure the spectrum and on the number of channels used.
Frequency of Occurrence of Phenomena	**Is Continuous Coverage A Requirement**	**Do We Need Detailed Data or Just Trends**	**Number of Ranges Required**	**Relationship to Other Measurements**	**What is Priority of Measurement? Can it be Dropped to Allow More Detailed Data from Another Experiment in Event of Flare, etc.?**
Not applicable One can for systems design purpose assume that 6 planetary and 6 comparison samples will be analyzed.	No	Detailed	2	Related to other measurements intended to establish chemical and mineral composition (neutron activation, X-ray diffractometer, gas chromatography, etc.	Yes, it can be dropped. However, once a measurement cycle has been started, it must not be interrupted The length of this cycle is. at present unknown.

c. Analysis of Mineral Composition

The purpose of the X-ray diffractometer is to produce X-ray diffraction patterns or signals equivalent to these for samples of planetary crust materials. These patterns are unique for each mineral. Consequently, the instrument is ideally suited for mineralogic analysis, that is, the identification of types of minerals in a sample, determination of relative abundances of the mineral types, and, finally, analysis of mineral composition of a complex mineral. In addition, mineralogical analysis provides lower limits on the amount of elemental constituents.

The basic principles of the instrument are shown in the following sketch.



The design is a conventional one but physical implementation requires utmost ultra-miniaturization.

The usefulness of this instrument depends on the availability of standard comparison patterns ~~such as contained in the ASTM~~ X-ray powder data file and ones ability to recognize similar patterns in the measured spectrum.

The practical operation requires that a crust sampler and a sample processor be available.

The specimen must be ground so that the maximum crystallite size does not exceed a value that limits the random orientation of powder components, nor is so small that pattern is unrecognizable. The range in crystallite size should fall toward the high end of the 75-500 A^{o} range. The radiation source is a miniature X-ray tube employing copper target. The 25-kv tube can currently be reduced in size to about 5 in. in length and 13 oz in weight. The detector is a proportional Geiger Counter. It is mounted on a goniometer which in turn is geared to a sample holder in such a manner that the angle relationship of Bragg's law is satisfied. The tentative characteristics of the instrument are given in Table A-2. Among important characteristics not shown is the peak signal-to-background-noise ratio which as a minimum should be 30:1.

In conclusion, it should be mentioned that one can conceive of a diffractometer operating free of sampler and sample processor. What would be involved is surfacing of the instrument and providing for the motion of both the source and the detector. However, such an approach may yield data which are more difficult to interpret.

d. Volatile Constituents in the Planetary Crust

Purpose

The purpose of this instrument is to analyze the volatile constituents of the planetary crust material. This information will provide partial answer to the chemical composition

TABLE A-2: SPECIFICATIONS FOR POSSIBLE INSTRUMENTATION FOR MINERALOGICAL ANALYSIS OF PLANETARY MATERIAL

<table>
<tr><th colspan="2">Experiment, Priority</th><th>Instrument</th><th>Special Requirements Such as Booms or Antennas</th><th>No. of Directional Channels, Stabilized Vehicle</th><th>Reference to Instrument Design</th></tr>
<tr><td colspan="2">Mineralogical analysis of planetary material</td><td rowspan="5">An ultraminiaturized X-ray diffraction instrument of conventional design.</td><td rowspan="5">Material sampler and sample processor needed for operation; material to be crushed so that crystallite sizes fall within the range of 75-500 Å</td><td>None</td><td rowspan="5">Any handbook, textbook, etc., on X-ray diffraction analysis.
Phillips Electronics Company work for JPL.</td></tr>
<tr><td colspan="2">Size</td><td rowspan="2">No. of Directional Channels, Spinning Vehicle</td></tr>
<tr><td colspan="2">10 x 10 x 10 inches exclusive of electronics and power supply</td></tr>
<tr><td>Average Bit Rate</td><td>Bits/event</td><td rowspan="2">None</td></tr>
<tr><td>Counter output at 3500 counts/sec. max. Goniometer output at 99 cps</td><td>50, 000</td></tr>
<tr><td>Weight</td><td>Power</td><th>Problem Areas</th><th>Hardware Development Status</th><th>Orbit Preference</th><th>Comments</th></tr>
<tr><td>15. 8-diffractometer
0. 5-electronics
8. 5-power supply
24. 8 lbs total</td><td>1, 700 watt-hours exclusive of heater power</td><td rowspan="3">Reliability of the miniaturized X-ray tube; operating life under extreme environmental conditions.
Sensitivity of performance to changes in power supply and ambient temperature.
Presence of high voltage (25 KV)</td><td rowspan="3">An instrument intended for use on Surveyor mission is under development by Phillips Electronics Company, Mount Vernon, N. Y., under contract with the Jet Propulsion Laboratory.</td><td rowspan="3">None</td><td rowspan="3">Energy requirement is based on the assumption that 6 planetary and 6 standard samples will be processed. The instrument requires 8 command channels.</td></tr>
<tr><td colspan="2">Temperature Restrictions For</td></tr>
<tr><td>Reliable Data

This remains to be studied.</td><td>Permanent Damage

This remains to be established.</td></tr>
</table>

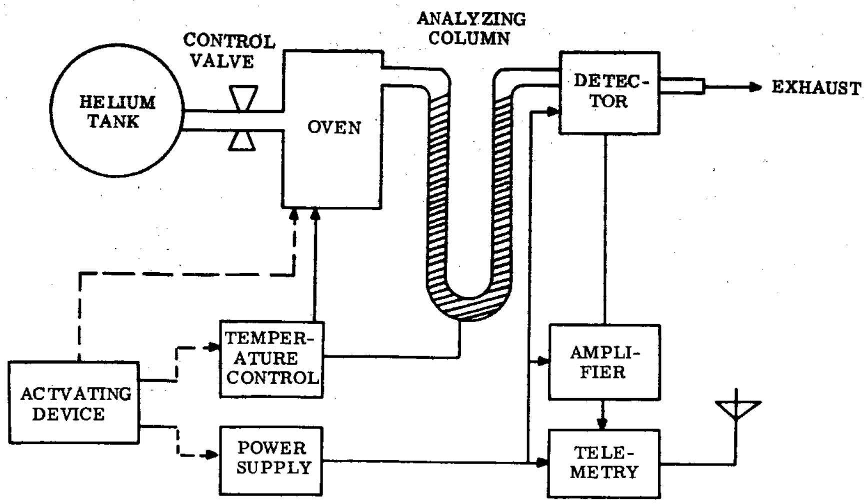
TABLE A-2 (CONT.)

Description of Transducer	Nature of Transducer Output (Analog or Digital)	Resolution Desired	Dynamic Range	Response Time (Fastest Expected)	Duration of Phenomena Being Measured
Conventional Proportional Geiger Counter which rotates about the axis located at the sample holder.	Ionization Pulses	Width of the ionization peak at 1/2 height, to be no larger than 2θ = 0.2 degree and at 1/10 height not to exceed 0.4°. Total scan region to cover range $7\text{-}180^{\circ}$ at rates of 1/2 and 4 degrees per minute.	30 to 3,500 counts per second	To handle 3,500 counts per second without distortion	At the slowest scan rate of 4°/min the complete arc will be covered in about 3/4 hour. This has nothing to do with the phenomenon itself. Initiation of a measurement cycle to be done by a timer or ground control.
Frequency of Occurrence of Phenomena	Is Continuous Coverage a Requirement?	Do We Need Detailed Data or Just Trends?	Number of Ranges Required	Relationship to Other Measurements	What is Priority of Measurement? Can it be Dropped to Allow More Detailed Data from Another Experiment in Event of Flare, etc.?
Not applicable	Yes, once a cycle has been started.	Detailed data	The entire dynamic range to be covered in one range if possible.	Rather indirectly lated to elemental as well as proximate analysis/neutron activation α-scattering, gas chromatography, etc.	It can be dropped at any time. However, once an analysis cycle has been started, it must not be interrupted.

of the surface layer. Selection of this instrument is based on its high sensitivity, relative simplicity, ruggedness, and rapid analytical capability over a wide range of constituents.

Basic Principles

The block diagram of a gas chromatograph is shown in the following sketch:



It is assumed that a sample of crushed planetary material is delivered to the instrument oven. Upon delivery of the sample the oven is sealed and heated to drive-off gaseous constituents in the sample. The unknown mixture is injected into an inert carrier gas (helium) in the form of a slug which is pushed through an analytical column containing solid support. Various gaseous constituents in the stream will have different affinities for the packing material. Due to adsorption or chemical interaction, each component of the solute is retarded in its progress through the column by an interval of time which is characteristic of the constituent. Thus, each component appears at the detector mounted at the effluent end after various lengths of time. These intervals are measured by an electrical signal produced by the detector.

The detection system for gases in either by thermal conductivity or for heavier constituents by glow discharge devices whose breakdown voltage is altered by a particular contaminant. The position of the resulting electrical pulses on the time axis identifies the constituents and the amplitude is a measure of the volume concentration.

The ability to detect a particular constituent, separation, speed of response, sensitivity, etc., depends on the characteristics of the bed material, column length and diameter, and the rate of gas flow. Typical packing materials are molecular sieves of various sizes, carbo wax on solid support, apiezon, fluoropack, fluorine pickrate, silicone oils, ucon polyglycol, etc. To detect constituents of widely differing nature, a number of parallel columns can be used.

Tentative Configuration of a Specific Instrument

Assume that it is desirable to detect fixed gases such as hydrogen, carbon monoxide, nitrogen, methane, oxygen, argon, water, and organic gases with molecular weight below 150. This can be done with a 7-foot molecular sieve 5A column, a 15-foot carbowax 1540 on T-6 Teflon particle column, and a 12-foot column with Apiezon L-Carbowax 20 M-phosphoric acid on chromosorb support. This arrangement will not detect CO_2. The instrument should be capable of conducting about 20 complete analytical cycles each lasting about 100 minutes. Approximately 30 constituents should be resolvable. The sensitivity of the chromatograph is expected to be of the order of 10^{-10} mole for fixed gases and 10^{-12} mole for organic vapors.

Detector

Karmen detector can be used operating in the voltage breakdown region. This detector is insensitive to changes in applied potential, suffers minimal difficulties due to grounding, and does not require high purity helium for detection of fixed gases provided sufficient pressure exists at its input. The dynamic range of the detector is high. With suitable attenuation, a range as high as 10^4 is not unreasonable.

Electronics

The detector outputs are applied to chopper stabilized electrometer amplifiers. Differential output is recommended to avoid the effects of drift in power or of changes in the ambient temperature.

More detailed characteristics of the instrument are given in Table A-3.

References

Oyama, V. I., Wilson, E. M., Application of Gas Chromatography to the Analyses of Organics, H_2O, and Adsorbed Gases in the Lunar Crust

Weber, T. B., Monitoring of Moon Base Atmospheres by Gas Chromatography, Lectures in Aerospace Medicine, Jan. 8-12, 1962, School of Aerospace Medicine

e. Petrographic Microscope

General Remarks

The texture and mineralogical composition of a particular rock results directly from the origin of the rock. Consequently, study of the above items provides clues to the rock's origin provided they can be interpreted. There is a vast reservoir of skill in such interpretations among terrestrial petrologists, particularly in studies involving thin sections of the rock and transmitted light. However, such work requires extensive preparation of samples which is impractical to conduct automatically. Therefore, operation on loose or crushed samples remains as the only alternative. However, usefulness of such an approach is limited because there is little previous experience in interpreting information obtained. This statement also applies to work with the reflected light. In view of the above remarks, the subsequent discussion indicates merely a crude approach to a possible experiment. A great deal more thought will be required before one can recommend such an experiment.

Purpose

It is hoped that the examination of crushed rock samples in transmitted light will accomplish the following:

(a) Identify the rock type and obtain an approximate mineralogical composition

(b) Detect glasses (cross-polarized light needed) and estimate their approximate composition from their refractive indices

(c) Determine the shape and size distribution of particles of loose material

Schematic Description of the Instrument

A petrographic microscope will consist of three parts:

TABLE A-3: SPECIFICATIONS FOR INSTRUMENTATION FOR IDENTIFICATION OF VOLATILE CONSTITUENTS OF THE PLANETARY CRUST

<table>
<tr><th colspan="2">Experiment, Priority</th><th>Instrument</th><th>Special Requirements Such as Booms or Antennas</th><th>No. of Directional Channels, Stabilized Vehicle</th><th>Reference to Instrument Design</th></tr>
<tr><td colspan="2">Identification of volatile constituents of the planetary crust</td><td rowspan="5">Gas chromatograph capable of about 20 analytical cycles with 30 constituents to be separated.</td><td rowspan="5">Surface sampler and sample processor (e.g., crushing) are needed for operation.</td><td rowspan="2">None</td><td rowspan="5">Beckman Instruments Incorporated under contract to Jet Propulsion Laboratory (Lunar and Planetary Exploration Colloquium, Vol. 3, No. 2, 1963, North American Aviation Incorporated).</td></tr>
<tr><td colspan="2">Size</td></tr>
<tr><td colspan="2">8 x 8 x 10 inches</td><td>No. of Directional Channels, Spinning Vehicle</td></tr>
<tr><td>Average Bit Rate</td><td>Bits/event</td><td rowspan="2">None</td></tr>
<tr><td>13</td><td>80,000</td></tr>
<tr><td>Weight</td><td>Power</td><td>Problem Areas</td><td>Hardware Development Status</td><td>Orbit Preference</td><td>Comments</td></tr>
<tr><td>14 lbs</td><td>12 watts</td><td rowspan="4">Helium leakage from the pneumatic system.
Temperature Control of Column Package (to be held at about 100^{o}C).
Mechanism sealing the oven and dumping the spent sample.</td><td rowspan="4">A flyable instrument is under development by Beckman Instruments Incorporated.</td><td rowspan="4">Not applicable</td><td rowspan="4">The instrument described is intended for use on Surveyor type vehicles. For Voyager its thermal design must be re-evaluated. This also applies to design of columns since different constituents may be of interest.</td></tr>
<tr><td colspan="2">Temperature Restrictions For</td></tr>
<tr><td>Reliable Data</td><td>Permanent Damage</td></tr>
<tr><td>Within range -30^{o} to 100^{o}C</td><td>Outside the range of -200^{o} to 100^{o}C</td></tr>
</table>

TABLE A-3 (CONT.)

Description of Transducer	Nature of Transducer Output (Analog or Digital)	Resolution Desired	Dynamic Range	Response Time (Fastest Expected)	Duration of Phenomena Being Measured
Karmen type detector operating as a glow discharge device. Electrically acts as a gas filled voltage regulator tube. Breakdown voltage changes in response to presence of a contaminant in carrier gas.	Analog 6 mv to 60 volts	Resolution in amplitude 1 part in 10^4 Separability of constituents. 30 constituents over a time base of 100 minutes.	10^4	0.1 second	Depends on the constituents, columns, rate of flow, number of constituents, etc. For 30 constituents and columns as specified, about 100 minutes.
Frequency of Occurrence of Phenomena	**Is Continuous Coverage a Requirement?**	**Do We Need Detailed Data Or Just Trends?**	**Number of Ranges Required**	**Relationship to Other Measurements**	**What is Priority of Measurement? Can it be Dropped to Allow More Detailed Data from Another Experiment in Event of Flare, etc.?**
No applicable. A total of 20 complete cycles available. Initiation of a cycle to be controlled by a timer or ground control.	No	Detailed Data	Two ranges desirable. Attenuation of about 200 to be provided. Attenuation to be variable in steps	Chemical composition measurement related to neutron activation α-scattering, X-ray spectrometer and similar measurements	No particular priority. However, once a cycle is started it should not be interrupted until completed.

(a) The microscope itself (optics) - with the associated focusing mechanism

(b) Sample processor and the associated means of presenting the sample to the microscope

(c) A TV monitor

The microscope optics can be of conventional design. The focusing mechanism may be controlled by a three position solenoid. These three positions correspond to the plane of best focus and to the plane below and above the plane of correct focus. This may be necessary to achieve at least one good image, and it permits observation of the direction of motion of Becke lines. The latter information supplies an estimate of the index of refraction of individual glass particles.

The required optical resolution must be sufficient to permit resolution of shapes of 5-10 micron particles. Also, the sample illumination system must be capable of providing plane and cross polarized light.

A sequence of images formed in the focal plane of the instrument is then relayed to the face plate of a vidicon tube. The overall design of the TV monitor may follow the same practice as currently envisioned for slow-scan space work. The resolution of the entire system will, of course, be limited by the dimensions of the vidicon electron beam. The latter is expected to be of the order of 25 microns in diameter.

It must be realized that any image forming device, even of relatively low image quality, requires transmission of an immense amount of information. Assuming that a resolution of 200 by 200 lines is needed and that about 8 shades of gray are desired, an individual frame will consist of 120,000 bits of information. Much of this information may be of no value, for instance, image of the interfering background, etc. Therefore some pre-programmed discrimination may be necessary.

Another serious problem is associated with the preparation of a sample for microscopic examination. A suggestion has been made that crushed material be delivered to a centrifugal device, heated, and then thrown against a thermoplastic tape driven by a rotary solenoid. Undoubtedly other systems could be devised.

Physical configuration of the entire instrument and its power consumption could be estimated as follows:

	WEIGHT (LBS)	DIMENSIONS (INCHES)	POWER (WATTS)
Microscope	15	12 x 12 x 4	8
TV Monitor	15	6 dia. x 12 length	8

As far as is known, no prototypes of such an instrument have been developed. However, feasibility studies have been conducted by the Armour Research Foundation for the Jet Propulsion Laboratory.

f. Measurement of Mechanical Properties of Planetary Soils

The purpose of this measurement is to establish reasonably accurate values of the load bearing strength and shear strength of planetary soils. These measurements will also indicate something about parameters such as cohesive and frictional moduli, bearing stability, etc. Knowledge of these properties is of interest in designing subsequent experiments and indirectly contributes to the interpretation of geophysical measurements of other bulk properties of the planetary material.

Measurement Principles

(a) Load Bearing Strength

Load bearing strength can be obtained by measuring penetration of a footing of a given area into the soil under the action of a given load. It is well known that bearing strength is a function of the footing area. Consequently, it is necessary to measure the load bearing strength for a number of reference footing areas.

(b) Shear Strength

Shear strength can be deduced from measurement of the angular displacement of a reference footing subjected to a given torque and normal applied force. This follows from the fact that shear strength can be represented as a linear function of the normal stress. The slope of this relationship is measured by the displacement angle, and the intercept along the shear strength axis is equal to the cohesion coefficient of the material.

Tentative Configuration of a Soil Mechanics Instrument

The instrument may consist of two axially loaded, circular flat plates of different areas for the measurement of load bearing strength and one spudded ring to determine the shear strength. The whole assembly is to be lowered from the main vehicle. Each of the penetration units has a tubular housing containing a motor driven ball-screw actuator which transforms motor motion into a linear motion driving the plate. The applied load is measured by a semiconductor strain gauge and the displacement is monitored by a 10 turn potentiometer. Measurement for two reference areas hopefully provides a scaling factor for design purposes. The range of strengths measurable is restricted by the limitations on load and torque reactions available from the vehicle.

The design of the shear strength component is the same in every respect except that a torque is provided to rotate the spudded ring as it is being driven into the soil. The shear strength must be measured at several axial loads in order to define the linear relation between the shear strength and the normal stress. The applied torque is measured by a split-ring strain gauge. As before, the maximum torque is restricted to avoid excessive torque reaction on the vehicle.

It is meaningless to discuss the range of measured strengths unless one knows something about the weight and configuration of the vehicle. More detailed characteristics of a possible soil mechanics instrument are based on an arbitrary assumption that maximum measurable bearing strength is 20 psi and maximum friction angle is 120°. Further information on a possible experiment is included as Table A-4.

References

Thorman, H. Call,"Review of Techniques for Measuring Rock and Soil Strength Properties at the Surface of the Moon," Paper presented at the Automotive Engineering Congress, Detroit, Jan. 14-18, 1963 (SAE paper 632C).

g. Borehole Drill

In order to measure bulk properties of the planetary subsurface layers, it is necessary to provide an instrument for drilling a borehole. Two types of drilling are possible. In one, an attempt is made to obtain an undisturbed core sample. The reason it must be undisturbed is the fact that the bulk properties of samples are extremely sensitive to changes in the environment. However, drilling of this nature under dry cutting conditions is difficult and for this reason is not recommended for early exploration. Another type of drilling which avoids lubrication problems is percussion drilling. It can drill holes but is completely useless as a sampling device because it destroys the environment of the sample. Samples obtained by this method can be used only for compositional analysis.

TABLE A-4: SPECIFICATIONS FOR POSSIBLE INSTRUMENTATION FOR DETERMINING MECHANICAL PROPERTIES OF PLANETARY SOILS

<table>
<tr><th colspan="2">Experiment, Priority</th><th>Instrument</th><th>Special Requirements Such as Booms or Antennas</th><th>No. of Directional Channels, Stabilized Vehicle</th><th>Reference to Instrument Design</th></tr>
<tr><td colspan="2">Mechanical Properties of Planetary Soils</td><td rowspan="5">Instrument based on Bekker's Method of Trafficability Analysis. It consists of two penetration test units and one shear strength unit. Each unit employs flat circular plates or rings of known area to determine desired parameters.</td><td rowspan="5">Provisions must be made for lowering the instrument from its stowed position in the vehicle to a suitable position on the planetary surface.</td><td>Not applicable</td><td rowspan="5">1. Bekker, M.G., of the Road Locomotion, Univ. of Mich., 1960

2. General Motors Defense Laboratory Reports on their work on contract to JPL</td></tr>
<tr><td colspan="2">Size</td><td rowspan="2">No. of Directional Channels, Spinning Vehicle</td></tr>
<tr><td colspan="2">25" high, approximately 9" in diameter
Fairly open structure</td></tr>
<tr><td>Average Bit Rate</td><td>Bits/event</td><td rowspan="2">Not applicable</td></tr>
<tr><td rowspan="2">At present, not important</td><td rowspan="2">94</td></tr>
<tr><td>Problem Areas</td><td>Hardware Development Status</td><td>Orbit Preference</td><td>Comments</td></tr>
<tr><td>Weight</td><td>Power</td><td rowspan="6">No serious problems are expected.</td><td rowspan="6">An instrument suitable for use in an Earth environment has been under developed by General Motors Defense Lab. since 1961 (On contract to JPL)</td><td rowspan="6">Not applicable</td><td rowspan="6">Power requirement will be determined by the number of measurement cycles desired.
The range of strength values measurable will depend on the local weight of the vehicle and its configuration, that is on the limitations on load and torque reactions available from the parent vehicle.
In one measurement cycle approximately 13 quantities to be measured (5 axial loads, 5 penetrations, 3 angle displacements).</td></tr>
<tr><td>Instrument 9 lb
Electronics 5 lb
total - 14 lb</td><td>Tentative total energy required 1.5 watt-hrs</td></tr>
<tr><td colspan="2">Temperature Restrictions For</td></tr>
<tr><td>Reliable Data</td><td>Permanent Damage</td></tr>
<tr><td>To be determined</td><td>To be determined</td></tr>
</table>

TABLE A-4 (CONT.)

Description of Transducer	Nature of Transducer Output (Analog or Digital)	Resolution Desired	Dynamic Range	**Response Time (Fastest Expected)**	**Duration of Phenomena Being Measured**
Axial Loads: semiconductor strain-guage mounted on proving rings. Torques: A split ring strain-gauge Linear and Angular Displacements: Multiturn potentiometers	All Analog	0.5 lb in axial load, 0.1 inch in linear displacement, 0.5 degree in angular displacement	0-50 lb in axial load 0-8 inch in linar displacement 0^{o}-120^{o} in angular displacement	**Times normally applicable to strain-gauge bridge work. Essentially, the requirement is for slow response whose exact value is immaterial.**	**Not applicable**
Frequency of Occurrence of Phenomena	Is Continuous Coverage a Requirement?	Do We Need Detailed Data or Just Trends?	Number of Ranges Required	**Relationship to Other Measurements**	**What is Priority of Measurement? Can it be Dropped to Allow More Detailed Data from Another Experiment in Event of Flare, etc.?**
Test performed under control of a timer or in response to ground command.	No	Detailed	A single range for each linear and angular displacement and applied axial load	Indirect relation to the measurement of bulk properties of the material (electrical and heat conductivities, elastic constants, etc.)	**It can be dropped. However, the measurement must not be stopped in the middle of a measurement cycle.**

However, two useful purposes can be served by such a drill. It can provide experimental information on the strength of subsurface material by measuring the rate of penetration as a function of operating parameters. Also, the resulting hole can be used for a well logging type instrument to measure certain bulk properties of the surrounding material.

Instrumentation

A possible drill configuration would consist of a 600 watt electromechanical percussion and rotation drill with a shaft 5 feet long and 1.25 inches in diameter. The shaft is hollow and it contains a chamber which accepts drill cuttings. These cuttings are in effect a sample employed in other work. The drilling bit must have holes which can pass the cuttings into the shaft. The chamber volume is between 5-10 cubic inches. The drill will provide means for emptying the sample chamber. It is also desirable to program the operation so that the drill can be withdrawn from the hole in order to insert the well logging tool.

The loading force can be of the order of 50 pounds and it would be maintained by the drill feed mechanism. The exact magnitude of the loading force depends on the rock being drilled and on the gravity of the planet. The rock fracturing is primarily caused by the impact energy which is delivered to the bit by a percussion head. The magnitude and the rate of the impacts depend on experimental detail. Representative values may be 3.5 ft-lb at 3000 impacts/minute. Bit indexing and cuttings collection is accomplished by rotating the shaft at several hundred rpm.

The depth of the hole which can be drilled for the given expenditure of energy depends on the rock type.

It appears that the preliminary design should call for holes between 2 to 5 feet in depth corresponding to soft and hard rock types.

Since no lubricants and coolants are to be employed, the bit must of necessity be withdrawn from the hole at frequent intervals to allow cooling by radiation. This method of cooling is likely to be more efficient than resting the drill in the hole and depending on radiation to the hole wall.

It is appropriate to consider a design such that the drill and the well logging probe are combined in a single instrument. Such an instrument may weigh about 30 pounds. The drill penetration and the applied load will be monitored by suitable strain gauges and potentiometers. Further details are included as Table A-5.

References

1. Armour Research Foundation, "Lunar Drill Study Program," ARF 8208-6 Final Report, January, 1961, ASTIA Doc. AD 258618.

2. Hughes Tool Co., Final Technical Report on Feasibility Study of Drilling a Hole on the Moon, Houston, Texas, Sept. 1960, ASTIA Doc. AD258661.

3. Texaco, Inc., "Lunar Drill Feasibility Study," Final Report, Bellaire, Tex., January, 1961, ASTIA Doc. AD 258683.

h. Well Logging Tool

Measurement of bulk properties of the planetary material can be performed on the surface as well as subsurface materials. This discussion is concerned with the latter case. The

TABLE A-5: SPECIFICATIONS FOR EQUIPMENT FOR DRILLING INTO THE PLANETARY SURFACE MATERIAL

Experiment, Priority		Instrument	Special Requirements Such as Booms or Antennas	No. of Directional Channels, Stabilized Vehicle	Reference to Instrument Design
Surface Drilling		Electromechanical percussion and rotation drill using a tungsten carbide drill bit with holes	Must be set on the surface of the planet from the stowed position. To remain rigidly attached to the spacecraft.	Not applicable	Reports of Hughes Tool Co., Armour Research Foundation and Texaco, Inc., to JPL Documents AD-258661 AD-258618 AD-258683
Size				No. of Directional Channels, Spinning Vehicle	
5 ft long 5 inches in dia.				Not applicable	
Average Bit Rate	Bits/event				
Immaterial	To be maximized	Problem Areas	Hardware Development Status	Orbit Preference	Comments
Weight	Power	Cooling of the drill bit. Sample is disturbed and useless for physical measurements	Feasibility prototypes for lunar work developed by Hughes Tool Co., Texaco, Inc. and the Armour Research Foundation on contracts to JPL	Not applicable	Considerable development work needed primarily to reduce weight and power consumption. Development of undisturbed core drilling techniques needed.
30 lbs	600 watts (≈1000 WH)				
Temperature Restrictions For					
Reliable Data	Permanent Damage				
None	None				

TABLE A-5 (CONT.)

Description of Transducer	Nature of Transducer Output (Analog or Digital)	Resolution Desired	Dynamic Range	Response Time (Fastest Expected)	Duration of Phenomena
Position of drill bit and the load applied to be monitored by conventional strain gauges and multi-turn potentiometers	Analog	Drill to be withdrawn at intervals from 1 in. hard rock to 0.5 ft in softer material	0-5 ft	Not applicable	Not applicable
Frequency of Occurrence of Phenomena	**Is Continuous Coverage a Requirement?**	**Do We Need Detailed Data or Just Trends?**	**Number Ranges Required**	**Relationship to Other Measurements**	**What is Priority of Measurement? Can it be Dropped to Allow More Detailed Data from Another Experiment in Event of Flare, etc.?**
Not applicable. Drilling to be performed until energy is exhausted.	Not applicable	Not applicable	Not applicable	Related to experiments on soil properties. Also used as a sampling device for gas chromatograph and possibly for other compositional analyses.	Can be dropped any time

instrument used is a counterpart of well logging probes commonly used in geophysical explorations. The range of properties which one can measure is very large and, therefore, it is necessary to limit such an experiment to just a few reasonable measurements. The choice suggested in this discussion is not unique, and it is likely that other investigators would propose different measurements. All measurements will utilize the borehole resulting from the drilling operations.

Measurements

The following measurements may be of interest:

1. Acoustic Velocity

The primary purpose of this measurement is to ascertain elastic constants of the planetary material and their variation with depth. The experiment is basically another variant of seismic measurement described elsewhere in this discussion. The sensor used is an accelerometer in firm contact with the borehole wall. The compressional wave whose velocity is to be measured is initiated by explosive charges or hammer blows. In fact, the same source of energy can be used as for the active seismic experiment. Typically, such an instrument is capable of measuring the arrival times to several microseconds and velocities between 80-8000 meters/sec. The latter can be determined with an uncertainty between 10-20%.

2. Density

Density of subsurface material can most readily be measured by utilizing the back scattering of gamma rays. Such an instrument would utilize a collimated 40 mc Ir^{192} gamma ray source and a Geiger-Mueller counter separated from the source by a shield to prevent the detection of the direct radiation. The intensity of the scattered gamma rays is a function of the material electron density and consequently of the ratio Z/A (standard notation). Thus, by suitable calibration one can relate the received radiation to the overall density. The range of densities which is measurable is 0-7 gr/cm^3. Precision with which an individual value is obtainable depends on the nature of the material and on the value of density itself. One can expect uncertainties in the range of 5 to 20%.

3. Temperature

The most direct way to measure temperature and its variation in response to external heating is by placing a thermocouple in contact with the wall. The source of thermal energy can be a small filament light bulb. The bulb and the detector system must be thermally insulated from other parts of the tool by a system of shields or mirrors in order to simulate a "black body" cavity condition in the surrounding part of the hole. The difficulty with this method is that measurements may be affected by contact resistance in the thermocouple measurement. From the time-temperature history of the material, one can determine its thermal diffusivity. The range of this value which is encountered for geological materials is 10^{-2} to 10^{-6} cm^2/sec. If the temperature is measured to $\pm$ 4°K, the uncertainty in diffusivity will be about 50% of its true value.

A more satisfactory method of temperature measurement is to employ an interferometer spectrometer to measure radiation temperature. The instrument is basically a Michelson type two beam interferometer. Its output is the power density spectrum of the reflected plus emitted radiation in a suitable wavelength range (5 to 30 microns). This spectrum can be solved numerically for the equivalent black body temperature of the material. A highly miniaturized instrument of this type appears feasible.

4. Electrical Resistivity

All electrical properties of the planetary materials will be measured by standard techniques employing miniaturizing impedance bridges.

Thus, the resistivity can be determined from the measurement of the Q of a resonant tank circuit placed near the material. To achieve this, high frequencies are needed ($\approx 10^5$ to 10^6 cps). The quantity sensed is the eddy current losses in the measured material.

5. Relative Permittivity

This quantity is of great interest in defining reflectivity of the material at radio frequencies. The bridge arrangement would utilize a tuned circuit in parallel with a plate placed near the material. The relative permittivity is then determined by measuring the capacitance necessary to retune the circuit.

6. Magnetic Susceptibility

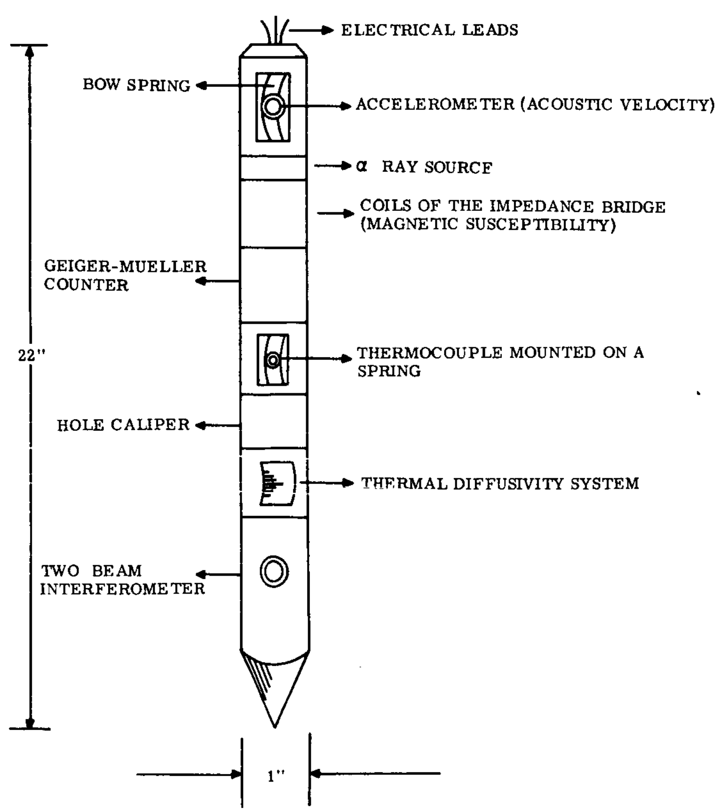
This measurement will utilize an impedance bridge with two coils of known mutual inductance. The presence of a material in the field of these coils changes their mutual inductance. The bridge is unbalanced by an amount which is a direct measure of the material's magnetic susceptibility. It is usual to measure this quantity in terms of the increment in resistance required to rebalance the bridge.

A possible range of magnetic susceptibility for terrestrial rocks is 10^{-4} to 10^{-1} in cgs units. The desired precision may be of the order of 10%.

The purpose of such a measurement would be to infer the content of free iron and other ferromagnetic materials in the planetary material.

Physical Configuration of the Instrument

An attempt can be made to combine all of the above measurements in a single instrument as shown in the following sketch.



Weight of the instrument is about 40 ounces.

Most instruments mentioned require no new technology. However, their miniaturization will require substantial amount of development work.

An attempt in this direction is currently underway at the Jet Propulsion Laboratory.

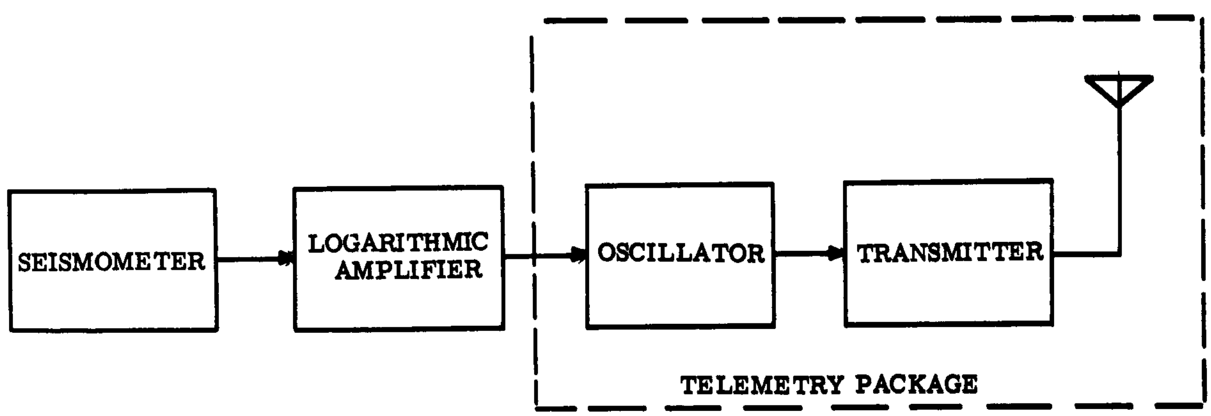
i. Passive Seismic Exploration of Planets

This note is primarily concerned with an instrument designed for early passive exploration of planets. A more sophisticated instrument is also mentioned. The instrument for early exploration is a single-axis, short-period seismometer intended for the measurement of microseismic activity which is an indication of thermal state of the planet and its tectonic activity. This is the simplest possible seismic experiment yielding a rather restricted amount of information. A considerably more sophisticated instrument is needed to measure surface and body waves as well as to conduct refraction and reflection measurements needed to ascertain gross structure of the planet.

Instrument

The heart of the instrument is the transducer consisting of a mass-magnet suspended by a coil spring which is restrained radially so that it responds only to motion parallel to its axis. The magnet has a circular gap which accepts a multiturn coil suspended from the frame of the instrument. The relative motion of the magnet with respect to the frame causes the magnetic field to be cut by the coil windings, thus generating an electrical signal which is proportional to the relative velocity of motion.

Basic block diagram of the system and a sketch of the transducer are shown below.



Some specific characteristics of a potentially useful instrument designed for lunar experiments are given in Table A-6. Changes in the instrument for work with other planets should not be difficult to make.

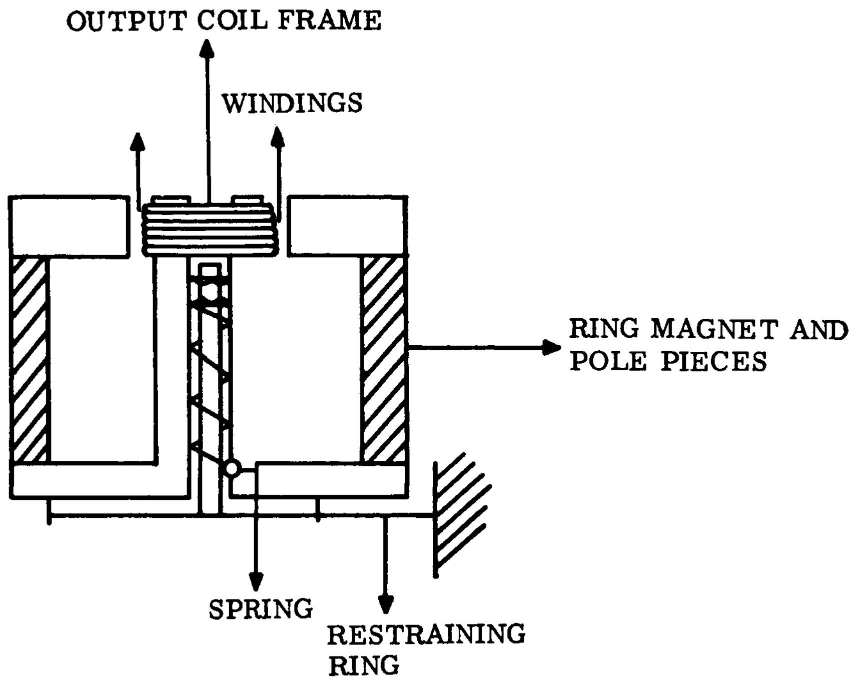
To conduct more detailed measurements, a three axis long-period seismograph may be needed. Basic design principles and all but the external characteristics of the instrument remain the same as for a single axis device. Weight and power requirements are increased by about a factor of 3. The projected size is approximately 15 inches by 10 inches in diameter. The sensing element may be of displacement capacitance type. The natural periods may be of the order of 5 seconds and the magnification about 10^4. Measurements with this instrument will indicate the planets' main structural features such as the existence of a crust, core, etc.

TABLE A-6: SPECIFICATIONS FOR POSSIBLE INSTRUMENTATION FOR MEASURING MICROSEISMIC ACTIVITY OF THE PLANETS

Experiment, Priority		Instrument	Special Requirements Such as Booms or Antennas	No. of Directional Channels, Stabilized Vehicle	Reference to Instrument Design
Microseismic Activity of Planets		Single Axis Seismometer using a moving coil velocity transducer. Elastic members of the transducer are protected by fluid from the effects of shock on impact.	The axis of the instrument must not deviate by more than 15^0 from the vertical. Provisions must be made for setting the seismometer on the ground.	Not applicable	Lehner, F.E., et al. Journal of Geophysical Research, Vol. 67, 4779, 1963 Reports of ITT Federal Laboratories to JPL & Lamont Geophysical Observatory on the lunar seismograph development
Size				No. of Directional Channels, Spinning Vehicle	
Seismometer: 5" x 4.5 dia. with Electronics: 16 x 7 in. dia.				Not applicable	
Average Bit Rate	Bits/event				
Corresponding to about 20 cycles/sec max.	Not applicable It is, however, desirable to obtain as many records as possible.	Problem Areas	Hardware Development Status	Orbit Preference	Comments
Weight	Power	Must be able to withstand relatively hard landing. Off-level tolerance must not exceed $\pm 15^0$. No serious problems foreseen at this time.	A prototype instrument developed by the California Institute of Technology (Seismic Laboratory) for JPL	None	On planets such as Mars and Venus a great deal of Microseismic activity may be caused by the atmosphere. Therefore, magnification of the instrument will be similar to that employed in the terrestrial seismometers rather than the value indicated in these sheets. Damping to be used is 0.7 to 1 of critical.
6-8 lbs	200 m w (oscillator plus amplifier)				
Temperature Restrictions For					
Reliable Data	Permanent Damage				
Should operate within the range $\pm 50^0C$ Some temperature control may be required.	Electronics range outside -30^0 to 100^0C				

TABLE A-6: (Cont.)

Description of Transducer	Nature of Transducer Output (Analog or Digital)	Resolution Desired	Dynamic Range	Response Time (Fastest Expected)	Duration of Phenomena Being Measured
Variable reluctance velocity transducer consisting of a ring magnet and a multiturn coil fixed to the instrument frame. Natural period is about 1 sec	Analog	Approximately 1 microvolt per millimicron of ground movement (peak to peak deflection)	30 db	If needed the high end can be extended to 300 cps. However, due to telemetry pass-band (which is 0.05-5 cps) the seismometer response need not exceed several tens of cycles (10-20)	Unknown
Frequency of Occurrence of Phenomena	**Is Continuous Coverage A Requirement?**	**Do We Need Detailed Data or Just Trends?**	**Number of Ranges Required**	**Relationship to Other Measurements**	**What is Priority of Measurement? Can it be Dropped to Allow More Detailed Data from Another Experiment in Event of Flare, etc.?**
Unknown	Yes	Detailed data	Logarithmic compression of data required	Closely related to measurement of surface and body waves by means of more elaborate seismometers as well as to the determination of elastic constants of ground material	Can be dropped at any time



References

Lehner, I. F. E., et al, Journal of Geophysical Research, Vol. 67, 47779, 1963.

j. Active Seismic Exploration of Planets

Active seismic experiment employs a net of detectors (geophones) and a controlled source of energy to obtain data which yield information on the local subsurface structure of the planet. The measurement is based on the fact that seismic waves are reflected from density discontinuities and that their velocity of propagation suffers dispersion which is characteristic of the density distribution with depth. Such measurements are of fundamental importance in determining local layering, existence of isostatic compensation of topographic features, possible locations of useful minerals, variation in crustal thickness, etc.

The difficulty with such an experiment is the fact that it requires considerable mobility in laying out an array of either energy sources or detectors or both over a considerable distance. Separation of the sources and detector must be accurately known. Seismic waves are best generated by setting off explosive charges. The recorded information consists of the distance between the source and the detector, detonation time, and the arrival time of the wave at the detector.

The detecting sensor can be a conventional moving coil geophone. It has been suggested that to lay out a reasonable net of charges or geophones a mortar-like device may be used. Using a propelling charge, this device can throw out a line with the attached source charges or geophones, thus distributing these into a linear array. The length of line which is deployed can be controlled. As far as is known, some preparatory work on this experiment was done at JPL.

As is to be expected the main concern was to develop deployment of the charges. An experimental mortar using a 1 gr charge was shown to deploy a 1 pound line to an equivalent distance of 1 mile at lunar gravity. This device also included automatic metering of the line.

Practical aspects of conducting such an experiment include selection of dimensions of the array, intensity of the energy sources, and perhaps choice of the source type. The latter fact is due to safety considerations.

Since practically no development work on such an experiment was done, discussion of specific instrumentation cannot be given.

k. Radar Propagation Experiments

Recent studies have indicated the versatility of radio propagation experiments in securing information about planetary, lunar and solar atmospheres, planetary and lunar surface features, and the interplanetary medium. This report describes the potential of the technique. The method has sometimes been called "bistatic radar," but the term "radio propagation experiment" is preferred because it more accurately describes the use of the technique for the Voyager mission. It is suggested that future references use the preferred terminology.

The principal feature of the radio propagation experiments is the involvement of three bodies. The earth serves as a platform for a powerful radar transmitter and a large tracking antenna. The object of study is extra-terrestrial, such as a planet or planetary atmosphere. The third body is a receiver on a space probe. A transmitter on the space probe relays information back to earth.

An example of the method is shown in Figure A-1. Radio (radar) pulses travel from the transmitter to the receiver on a space probe by the direct path and by a refracted or reflected path. A few or many frequencies may be transmitted simultaneously to analyze the frequency dependence of certain characteristics of the propagating medium and the planetary surface. Combining the two signals at the probe permits instrumental delay times and delays imposed on both signals by the interplanetary medium to be nulled out. Group and phase path length measurements may be utilized to determine the magnitude and time variation of the integrated electron density along both the direct and reflected paths. The direct path provides data on the interplanetary plasma. Passage of waves through the planetary ionosphere will permit the measurement of integrated ion densities. An abrupt change in reflectivity and range on the reflected path indicates a change in the level of reflection from the ground to the layer of maximum electron density in the ionosphere. The shift in differential path length and angle of reflection at which it occurs gives a measure of the height and density of the ionosphere maximum. Atmosopheric data is also derivable from these measurements.

An example may be noted. Assume that the ionosphere of Venus has a maximum electron density of 10^7 cm^{-3} at a given height above the surface and that a radio frequency of 37 mc/s is being used. When the angle of incidence for specular reflection to the space probe increases to about 40^o, the reflection point will rise abruptly from the planetary surface to the level of ionization. The consequent time delay along the reflected path will decrease several microseconds. The height of the layer and the electron distribution above the maximum can be determined. The use of several frequencies will permit simultaneous studies of different ionization levels.

Polarization and signal strength measurements may be used to deduce reflectivity and roughness parameters for the reflection points on the surface. The Brewster angle (at which only the horizontally polarized wave is reflected) may be determined with considerable accuracy to provide a measure of the average surface dielectric constant. Departures from the average would be indicative of the scale and distribution of undulations in the surface. If an average value of the dielectric constant may be found for a region, the average slope of the large scale irregularities may also be found. Fading surface signals must be averaged to compute the relative dielectric constant. The intensity and frequency

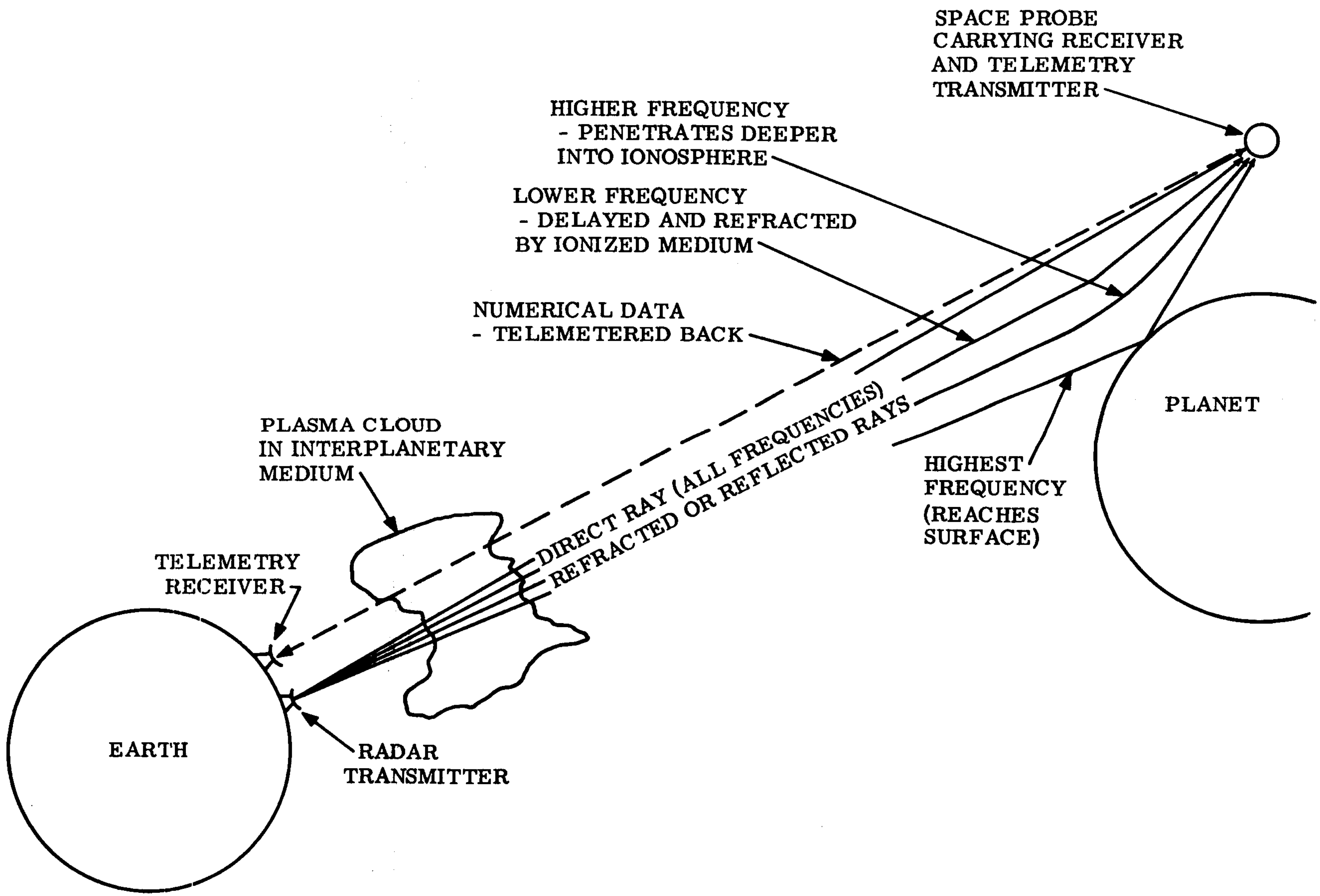


Figure A-1. Radio Propagation Experiment of fading are, however, indicative of large scale roughness parameters. Measurements of changes of reflectivity as a function of frequency make it possible, in principle, to determine the conductivity of the material and variations of dielectric constant and conductivity with depth. By measuring the reflectivity over a wide frequency range, it is theoretically possible to determine the average and the local conductivity of the surface material. Since various frequencies are sensitive to different depths, with long wavelengths penetrating the farthest, changes of conductivity with depth may theoretically be studied. A powerful earth transmitter is obviously required.

It is to be noted that specular reflection from the surface may be studied as a function of the angle of incidence (i). See Figure A-2. Monostatic radar techniques (transmitter and receiver at the same location) permit the study of perpendicular reflections ($i = 0^{\circ}$) only. The reception of both direct and reflected signals permits automatic calibration of the reflection coefficients. Monostatic radar requires the difficult determination of absolute system sensitivity in order to determine reflection coefficients.

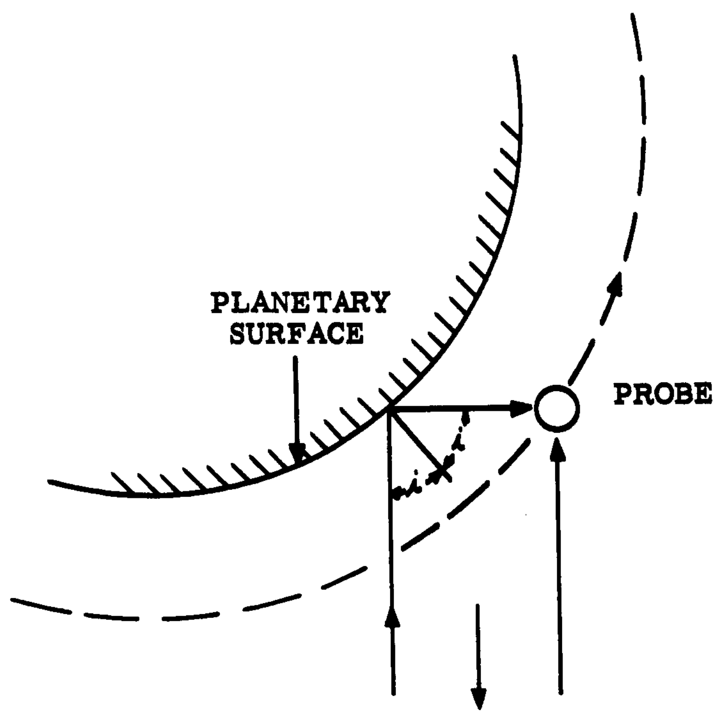


Figure A-2. Geometry of the Direct and Indirect Propagation Paths to a Voyager Orbiter

An excess of ionized gas in the near vicinity of the planet will refract and alter the phase velocity of radio waves. At or near grazing incidence ($i = 90^{\circ}$) these effects will be maximum. With a planetary Orbiter, two coherent, polarized, harmonically related signals will yield precision phase path measurements. It is apparent that optical (or more correctly, radio) path lengths are involved in these studies.

An example of a related effect was observed when the Crab Nebula was occulted by the moon. At a wavelength of 3.7 m, the nebula was occulted for 0.4 ± 0.26 minute longer for radio frequencies than for optical frequencies. If refraction was assumed to occur only on the sunlit side, an angular deviation of $13.4'' \pm 8.7''$ is implied. This suggests an electron density of several hundred per cubic centimeter near the lunar surface. However, the experimental data are not precise enough to determine the presence of a lunar atmosphere. Since the amount of refraction is proportional to λ^2, the use of relatively low frequencies is important. By using two frequencies, one of which is comparatively low (20 to 40 mc) and the other high (2000 mc), only the relative times of occultation (signal loss) need to be measured to determine the amount of refraction and the ionospheric density. The high frequency undergoes little or no refraction.

Either phase or group signals may be used to determine the departure from sphericity of the average planetary surface. This may be especially useful for Venus, whose surface cannot be seen optically from above its atmosphere. Additional data about the Venusian atmosphere will be desirable to evaluate the feasibility of this method. Range and range-rate data will help establish an accurate value for the planetary mass of Venus.

Another experiment which may be performed is the determination of the average interplanetary charged particle density between the earth and a space probe. Accurate measurements of time variations of this quantity would provide data about the steady solar wind and bursts associated with sunspots. One method of studying changes is based upon polarization measurements. If charged particles with high average velocities penetrate space, the stream of particles causes the medium to become anisotropic. The medium propagates one wave of linearly polarized radiation perpendicular to the net particle flow and another at right angles to the first wave. The polarization ellipse of a propagating wave will be changed as it traverses the medium. The change is due to the relativistic increase in mass of streaming electrons. It is functionally related to the radio frequency, the charged particle density, and the net particle velocity.

Another sensitive and more reliable method of determining changes in integrated electron density involves the comparison in a space probe of the receiver carrier frequency of, for example, 50 mc/s with the 1/8 subharmonic of a 400 mc/s carrier. It has been computed that changes in electron concentration of no more than an average of 0.005 electrons/cm^3 along the total path would be detectable at the maximum range using a currently available parabolic dish. This range is about 5×10^7 miles, which is approximately the minimum Earth-Mars distance. By modulating both carrier frequencies at a pair of closely spaced frequencies (such as 9.1 and 10.1 kc/s), phase shifts can be compared to permit the calculation of the total electron content to within about 0.4 electron per cc over a range of 5×10^7 miles.

In the unlikely event that a comet passed between the space probe receiver and earth, the change in integrated ion density could be studied through the cometary mass and tail.

Information concerning the density, structure, and dynamics of the solar corona may be obtained by the use of a Venus Orbiter when the planet is near opposition. Relatively low frequencies will probably be required. The accuracy and reliability of this method has not been established.

A source of error in the interplanetary electron density measurements is the electron density in the Earth's ionosphere. This may be measured or predicted to within ± 10% by Earth satellites, incoherent scatter, or ionosondes. The Earth's ionosphere would contribute an uncertainty of only 0.5 electron/cc to the average interplanetary electron density.

When the equipment is not being used in the bi-static mode, several operations can be performed in the monostatic mode. These include space probe tracking, telemetry and the study of strong echoes from the sun, moon, and planets. It has been suggested that the large steerable antenna beam would be important for studying magnetic field effects in the Earth's upper atmosphere. Ionic gyro resonances have been predicted in the scatter spectrum. If detectable, a "radar mass spectrometer" may be used to identify ionic species, electron densities, and electron and ion temperatures in the ionosphere and magnetosphere.

Although some of the above experiments are not included in the requirements for the Voyager mission, it is evident that the equipment can be utilized advantageously in other experiments.

Another use of this scientific approach, which will not be discussed in this report, was mentioned by recent visitors to the Voyager Project from the Autonetics Division of North American Aviation. They suggested to E.H. Stockhoff of GE-MSD the possibility of using the bi-static mode for vehicle guidance purposes.

It must be emphasized that no radio propagation experiments have been performed. The feasibility and accuracy of a few of the above experiments is yet to be determined. NASA has authorized its presence in a PIONEER mission late in 1964 or 1965. Two frequencies (50 and 400 mc/s) will be used to make studies of phase and group velocities and polarization and amplitude of signals. The transmitter-receiver is a 150-foot steerable dish at Stanford University which is tunable across the range of 20 to 60 mc/s. It employs a 300 kw, CW transmitter which can be expanded to 600 kw. A 30 kw, CW transmitter is currently available for the 400 mc/s band. The transmitter will work to 0.5 astronomical unit at present.

The radio propogation experiment on the PIONEER flight, which will perform only a few of the experiments mentioned in this report, weighs 5 pounds and requires 2 watts. It has been estimated that more versatile equipment aboard Voyager will weigh 13 pounds (including antenna) and require 2 watts.

The Solar Probe Study recently issued by GE-MSD suggests that radio propagation experiments should be included on the vehicle to determine integrated electron densities. One of the members of the Solar Probe Team, Dr. R.T. Frost, visited the principal proponent of radio propagation experiments, Dr. Von R. Eshleman, Co-director of the Stanford Center for Radar Astronomy, Stanford University, Stanford, California. The evaluation of the method led to its inclusion in the solar probe vehicle as one of the prime experiments.

Radio propagation equipment is being included on a few Venus and Mars Voyager missions.

Summary

The following analyses, subject to additional feasibility and accuracy studies, may be performed by the radio propagation technique.

1. Integrated electron densities along direct and reflected or refracted paths
2. Height and density of ionospheric maxima
3. Reflectivity and roughness parameters for reflection points on the planetary surface
4. Dielectric constant and conductivity of the surface
5. Range and range-rate data to determine accurate planetary masses (especially for Venus)
6. Ionospheric and atmospheric density, profile, and structure (from probe occultations)
7. Value and change of value of integrated ion density during periods of solar activity
8. Density, structure, and dynamics of the solar corona when Venus is at or near opposition.
9. When not used in the bi-static mode, the monostatic radar may be used for space probe tracking, telemetry, echoes from bodies in the solar system, and ionospheric and magnetospheric analyses of the Earth's upper atmosphere.

SECTION 2. PARAMETRIC SYSTEM PERFORMANCE

In order to determine the most optimum system design consistent with the scientific objectives of the Voyager program given in Section 1.1, it is necessary to consider the effects of the energy requirements for the various Mars and Venus opportunities between 1967 and 1975 on the size Orbiters and Landers and the orbits achievable. With the introduction of an orbiting module on planetary spacecraft, the choice of the transit trajectory (and hence system performance) is based upon a somewhat different criteria than for the case where only a fly-by or entry module is utilized. Therefore, it is necessary to analyze the various opportunities on a parametric basis so as to understand how the total mission capability would vary with the type of orbits and Orbiter and Lander sizes. These results would then be used to identify preliminary Voyager concepts which show promise of fulfilling an evolutionary science program with a minimum of system modifications for the various opportunities.

2.1 LAUNCH VEHICLE PERFORMANCE

The launch vehicle performance that was utilized in this study is given in Figure 2.1-1. This performance is in terms of the injected weight capability as a function of C_3(the hyperbolic excess velocity equared) for the Saturn CIB with an SVI upper stage and the Titan IIIC launch vehicles. For the purposes of the study it was assumed that the weight of the shroud and the adapter was not included in the payload capability. Therefore, this curve represents a maximum allowable weight for the Voyager Spacecraft.

Since this study was considered to be only a conceptual design of a Voyager Spacecraft, no allowance was made in the analyses for either possible spacecraft weight increases or possible degradation in booster performance between now and the time of launch. However, in Section 3.3 of this volume a brief analysis was conducted to indicate the influence of a 10% reduction in launch vehicle performance on spacecraft capability.

With this relationship between the weight capability of the various boosters and the energy requirement the next step was to determine the energy requirements for the various missions under consideration.

2.2 ORBIT INSERTION VELOCITY REQUIREMENTS

Since for most orbits of interest the Voyager Spacecraft will not approach the planet at the velocity required for a planetary orbit, a velocity change must be made. This velocity correction required is a function of the hyperbolic excess velocity of the spacecraft, the particular planet in question, and the final planetary orbit desired.

One factor that will place a restriction on the planetary orbit to be utilized is the question whether it is possible to sterilize the Orbiter. For the purposes of this study NASA has specified that if the minimum altitude for the Mars circular orbit is approximately 1000 n.mi. or more the Orbiter will not require sterilization (for a highly elliptical orbit the perifocus can be as low as 800 n.mi. before sterilization of the orbiter would be required). Therefore, the minimum altitude for all Mars orbits was set at 1000 n.mi. irrespective of whether it was circular or elliptical. This conservatism was employed because of guidance uncertainties which would necessitate biasing the aiming point.

The velocity required for orbit insertion is equal to the difference between the approach velocity and the perifocal velocity of the particular orbit. The approach velocity (V_a) is determined from the relationship;

$$V_a = \sqrt{V_h^2 + V_e^2}$$

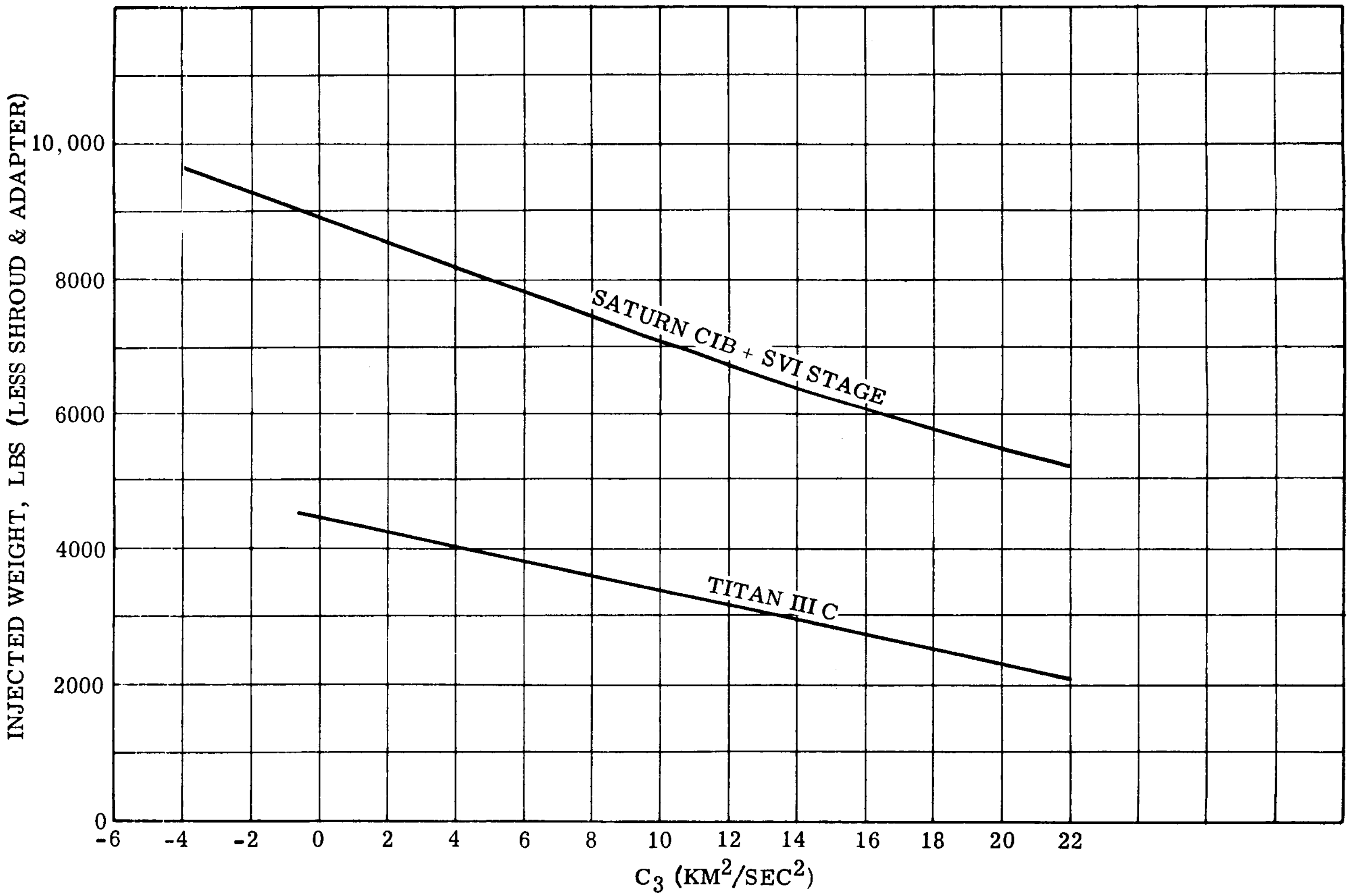


Figure 2.1-1. Launch Vehicle Performance

where V_h = hyperbolic excess velocity

V_e = planetary escape velocity at the altitude for orbit insertion.

while the perifocal velocity (V_p) is given by

$$V_p = \sqrt{g_s R^2 \left(\frac{2}{h_p + R} - \frac{1}{\frac{h_p + h_a}{2} + R} \right)}$$

where g_s = 12.56 ft/sec for Mars

g_s = 28.3 ft/sec for Venus

R = 1830 n.mi. for Mars

R = 3340 n.mi. for Venus

h_p = perifocal altitude

h_a = apifocal altitude

The resulting velocity curves required as a function of hyperbolic excess velocity and planetary orbit desired are given in Figures 2.2-1 and 2.2-2 for Mars and Venus respectively. It is to be noted that for a given hyperbolic excess velocity and desired orbit, the insertion requirement for Venus is significantly higher than for Mars due to the larger size of Venus. In addition, the hyperbolic excess velocity at arrival is generally (but not always) higher for Venus than Mars as will be seen in Section 2.6

2.3 ORBITAL PERIODS

Since the Orbiter may be utilized for such purposes as TV and radar mapping and as a communication relay for the Landers, the orbital period is of interest in determining the best overall orbit.

The orbital periods were therefore determined from the following equations:

Venus

$$T^2 = 2.2 \left(\frac{h_p + h_a + 2R}{2R} \right)^3$$

Mars

$$T^2 = 2.65 \left(\frac{h_p + h_a + 2R}{2R} \right)^3$$

These relationships are plotted in Figures 2.3-1 and 2.3-2 for Venus and Mars respectively.

2.4 ORBIT INSERTION PROPULSION CAPABILITY

In order to achieve the velocity changes indicated in the previous section, either atmospheric braking, a propulsion system or a combination of the two techniques could be used. It was determined early in the study that because of the atmospheric uncertainties for Mars and Venus and because of the guidance accuracy requirements, the use of atmospheric braking did not appear practical for the early Voyager missions. Therefore, only the use of proplusion systems were given further consideration.

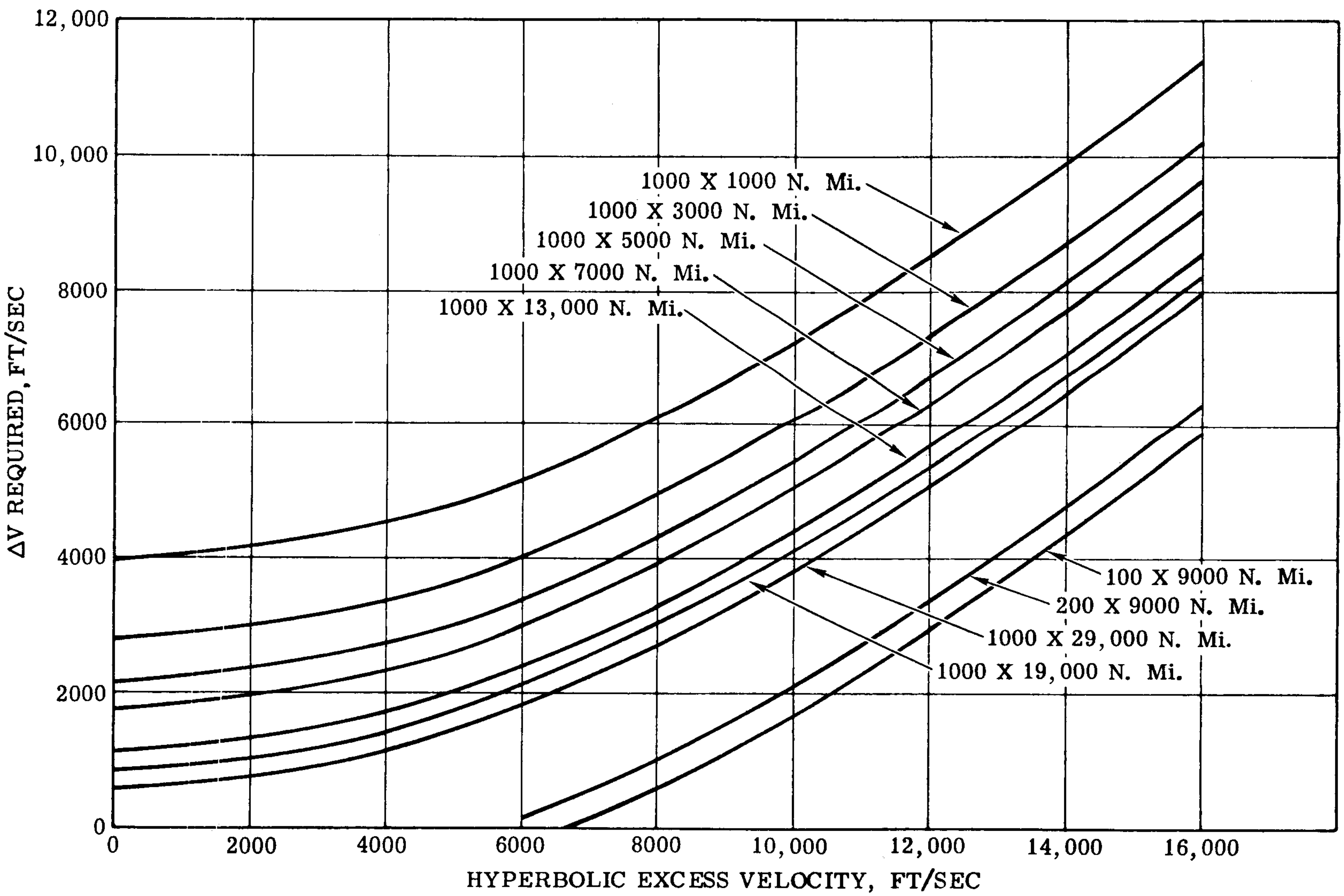


Figure 2.2-1. Martian Orbit Insertion Velocity Requirements

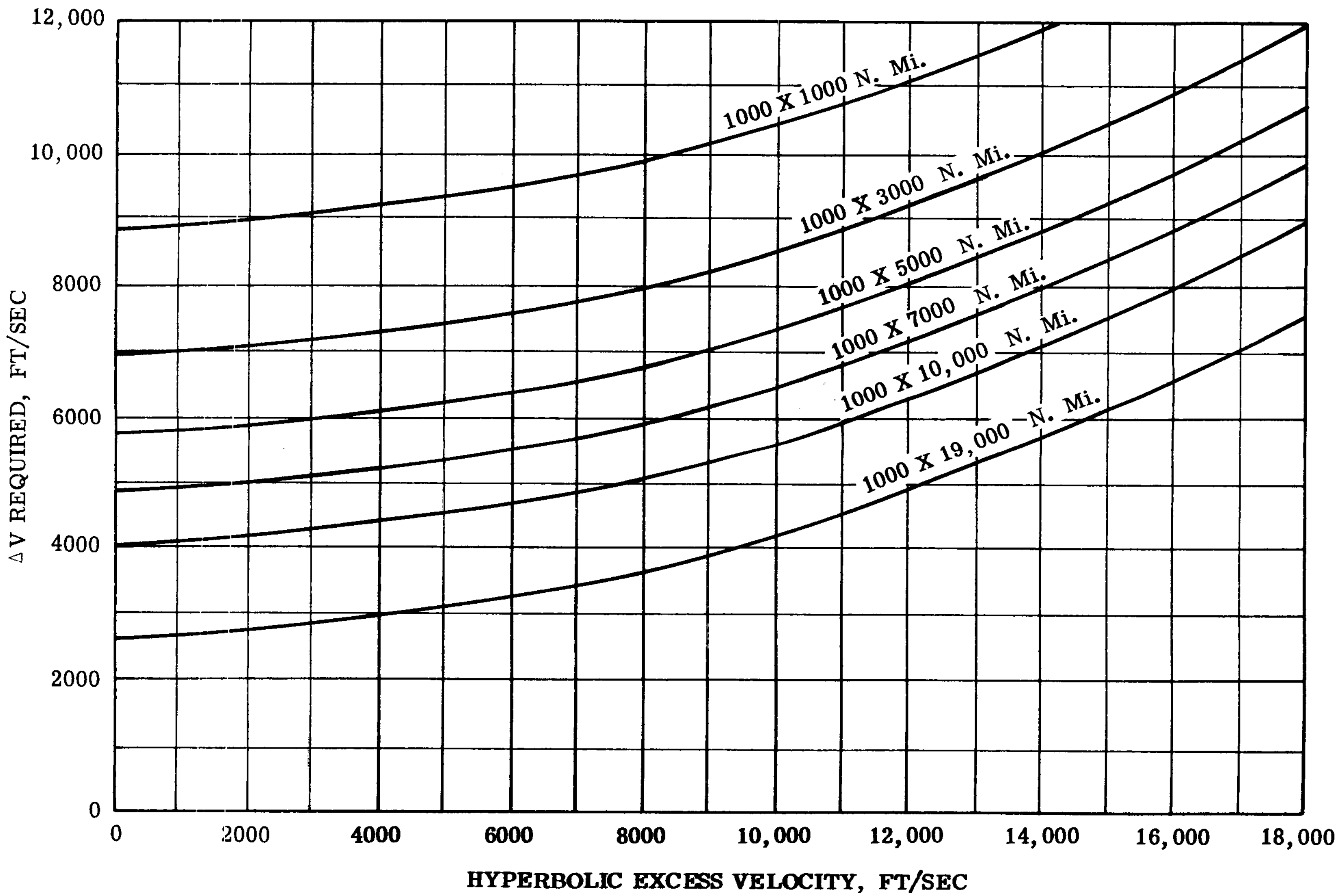


Figure 2.2-2. Venusian Orbit Insertion Velocity Requirements

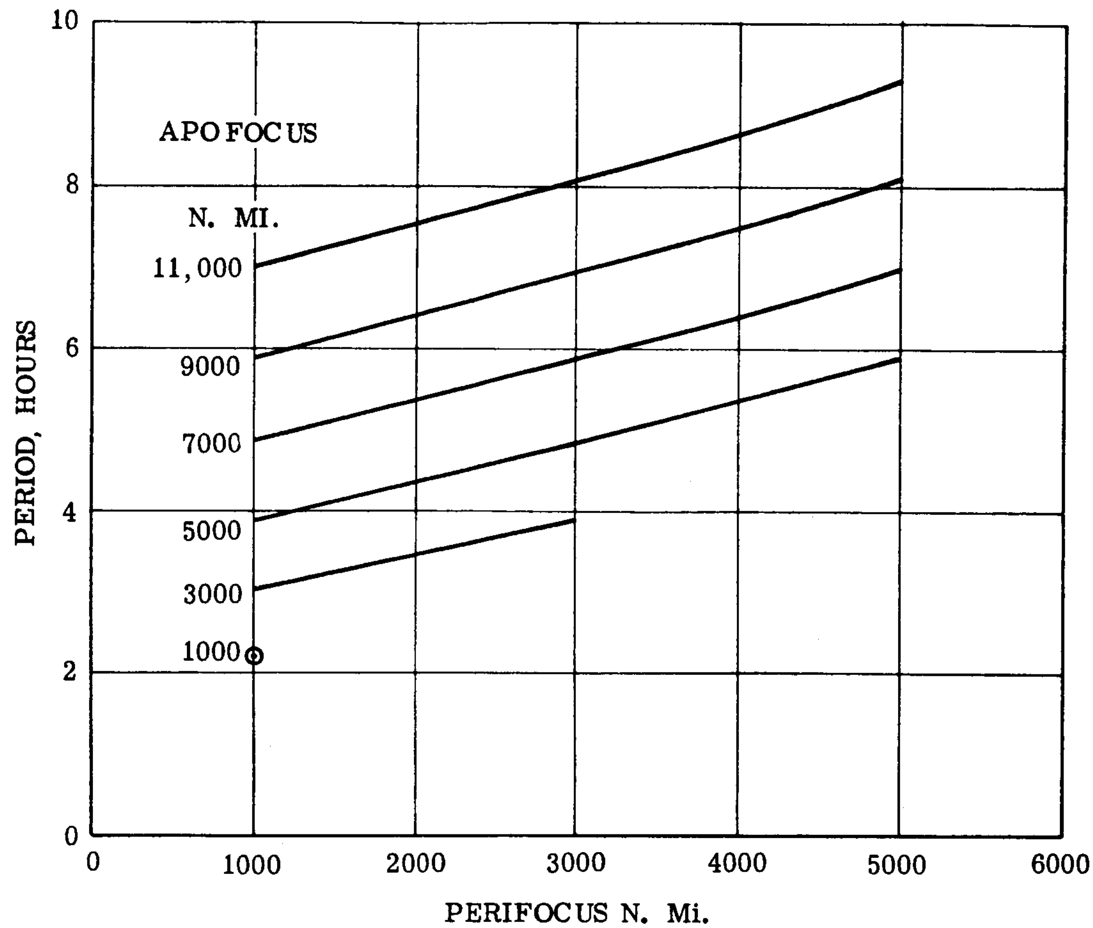


Figure 2.3-1. Orbit Period - Venus

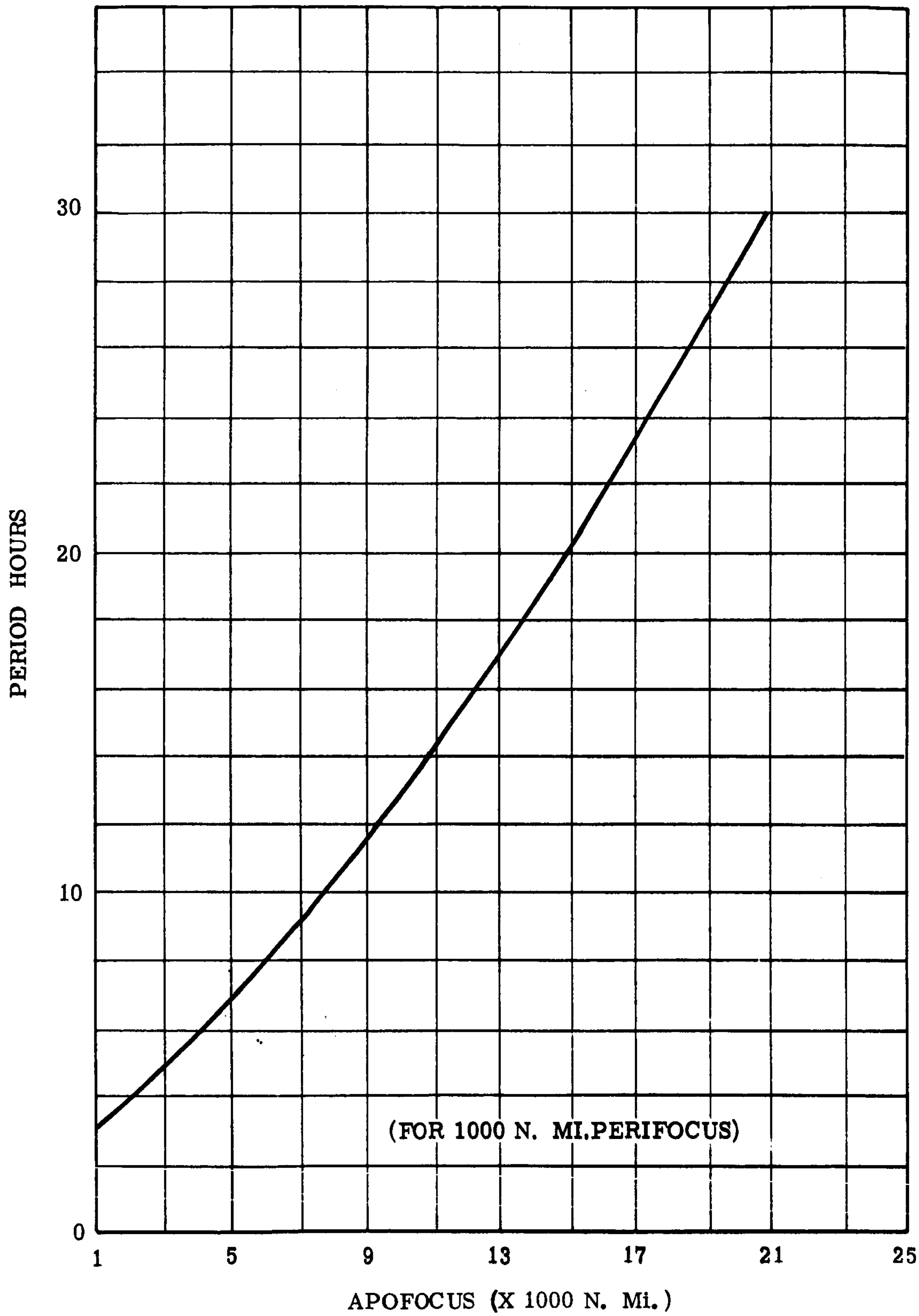


Figure 2.3-2. Orbit Period - Mars

As a first step in determining the approximate performance of Voyager systems on a parametric basis it is necessary to make an assumption as to the propulsion capability for the orbit insertion phase. It was assumed, therefore, that the propulsion system has a mass fraction of 0.85 and a specific impulse of 310 seconds. (These assumptions were made early in the study before the final selection of the propulsion system was made. However, the performance of the selected propulsion system is surprisingly close to these early assumptions.) Using these values it is possible to calculate the ratio of non-propulsive weight in orbit to the initial Orbiter weight as a function of propulsive velocity increment imparted. The resulting relationship is given in Figure 2.4-1.

2.5 THE RELATIONSHIP BETWEEN ORBITER AND LANDER WEIGHT

Utilizing the information in Sections 2.1 through 2.4 and the interplanetary trajectory characteristics given in References 1, 2 and 3, it is now possible to determine parametric system performance for the various opportunities.

For the purposes of this study a launch window of 30 days was considered satisfactory for all opportunities with the exception of Venus 1967. For this particular opportunity only one launch pad will be available. Therefore in order to obtain two launches during this opportunity, a launch window of 45 days was considered to be a necessity.

A first approximation of the desirability of the various opportunities from the standpoint of energy requirement (in terms of vis viva geocentric energy) can be had by considering Figures 2.5-1 and 2.5-2. The first figure indicates the absolute minimum energy for Mars opportunities between 1969 and 1977, while the second indicates the energy requirements for Venus opportunities from 1967 through 1975. Generally, the type I trajectories were favored for all opportunities since they are characterized by a shorter trip time. However, for a number of opportunities, noteably Mars 1969 and 1975, type II trajectories were utilized since the type I were unsatisfactory from the standpoint of range safety at AMR. For the purposes of this study, it was assumed that the launch azimuth restrictions were between 90° and 114°. This, when translated into declination of geocentric asymptope, placed a requirement that this parameter be between $+36^{\circ}$ and -36°.

From a consideration of injection energy requirements alone, the Mars 1969 and 1971 opportunities appear to be the best while the 1973 and 1975 opportunities are significantly poorer. For Venus the 1967 opportunity is the best with the energy requirements increasing with the years until 1975 when the cycle is repeated.

If the Voyager Spacecraft were to consist of only a fly-by or entry module the choice of transit trajectory would be easily ascertained by considering a 30 day launch window about the minimum energy trajectory and the desirability of the various opportunities could be readily evaluated from Figure 2.5-1 and 2.5-2. However, when an orbiting module is present which requires a relatively large ΔV for orbit insertion the approach velocity becomes a rather significant factor. Since in general the particular trajectory (a given launch date and trip time) that yields a minimum injected energy does not also yield a minimum arrival velocity, a wide range of launch data and trip times must be considered so as to maximize the sum of orbiting weight plus landing weight rather than maximizing injected weight alone.

Using this approach to obtain the most desirable launch date and trip time, it will be shown in later paragraphs that the performance possible during the Mars 1973 opportunity is on a par with 1969 (if an Orbiter is part of the mission) and not as bad as one would expect from a consideration of injection energy alone.

Since it was not known at the beginning of the study what combination of Lander and Orbiter weight would yield the most optimum system from the standpoint of scientific return, it was necessary to treat these values in a parametric fashion. Various combinations of

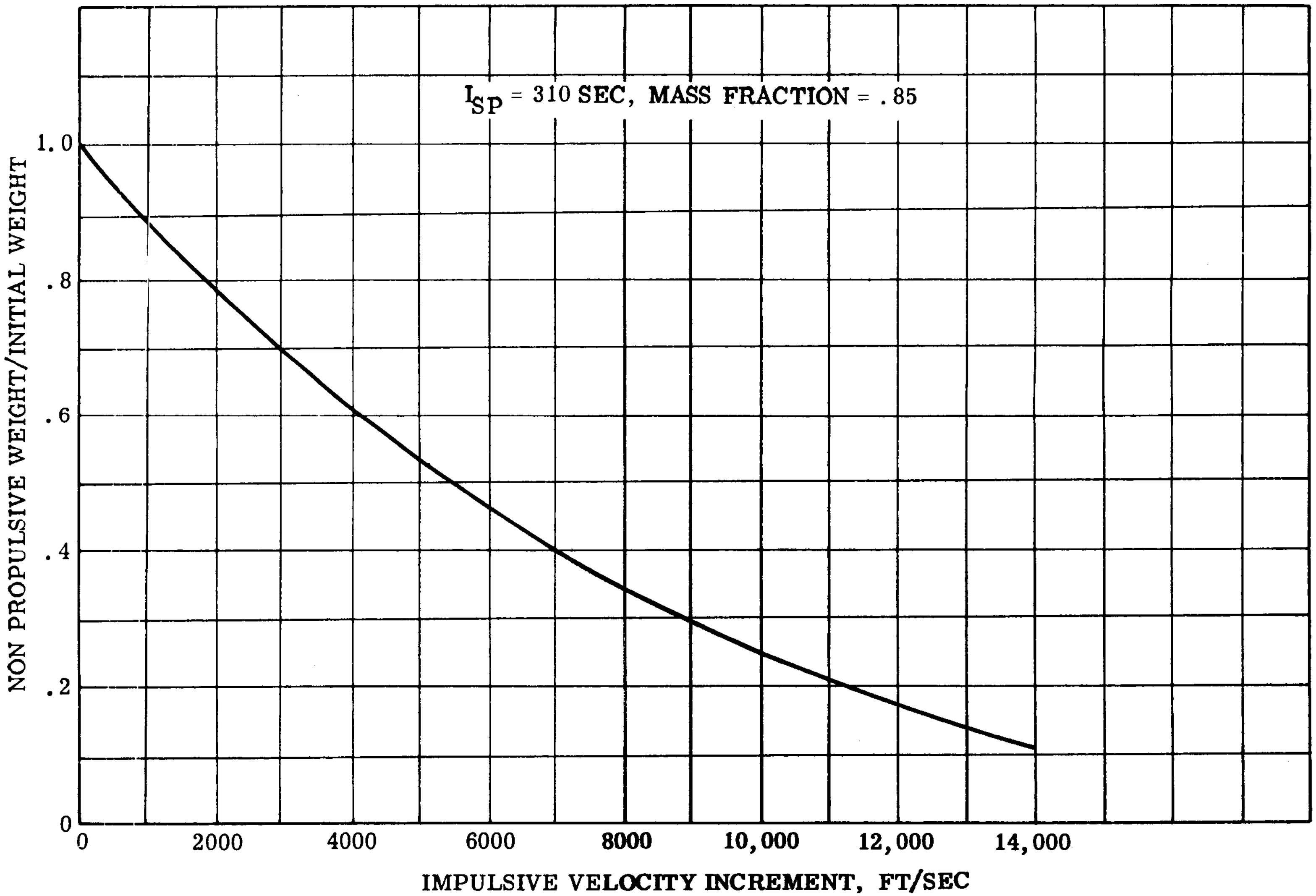


Figure 2.4-1. Orbit Insertion Propulsion Performance

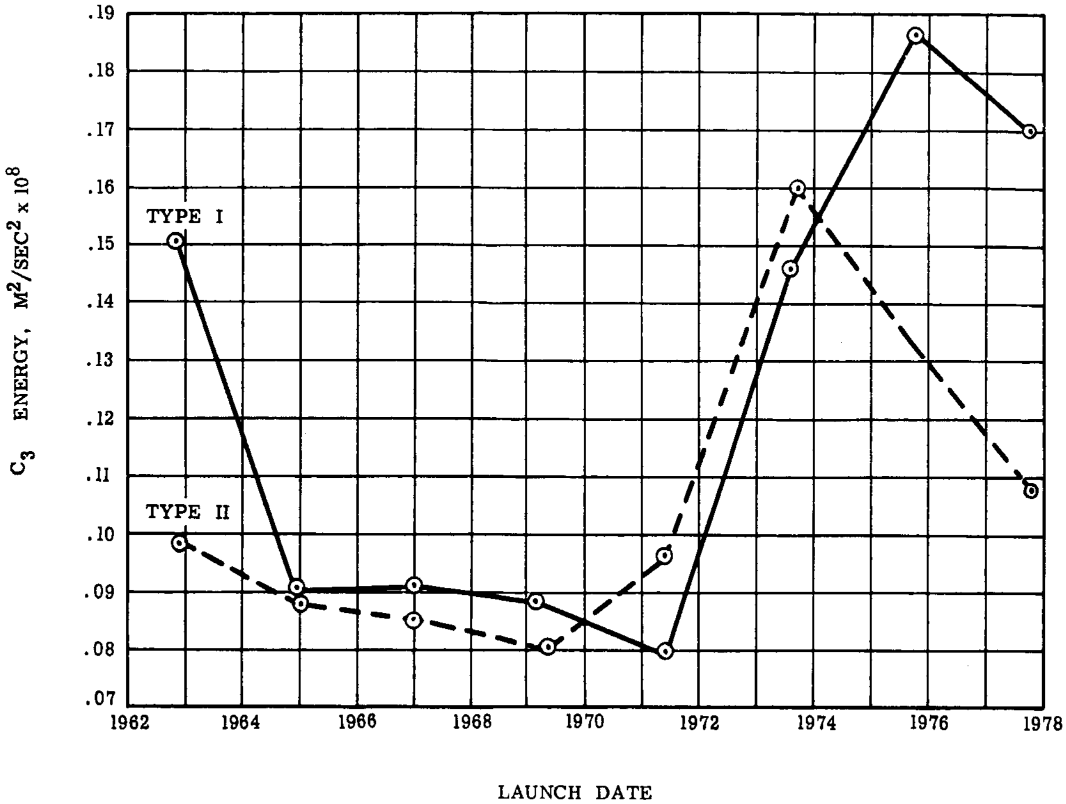


Figure 2.5-1. Absolute Minimum Energy for Mars

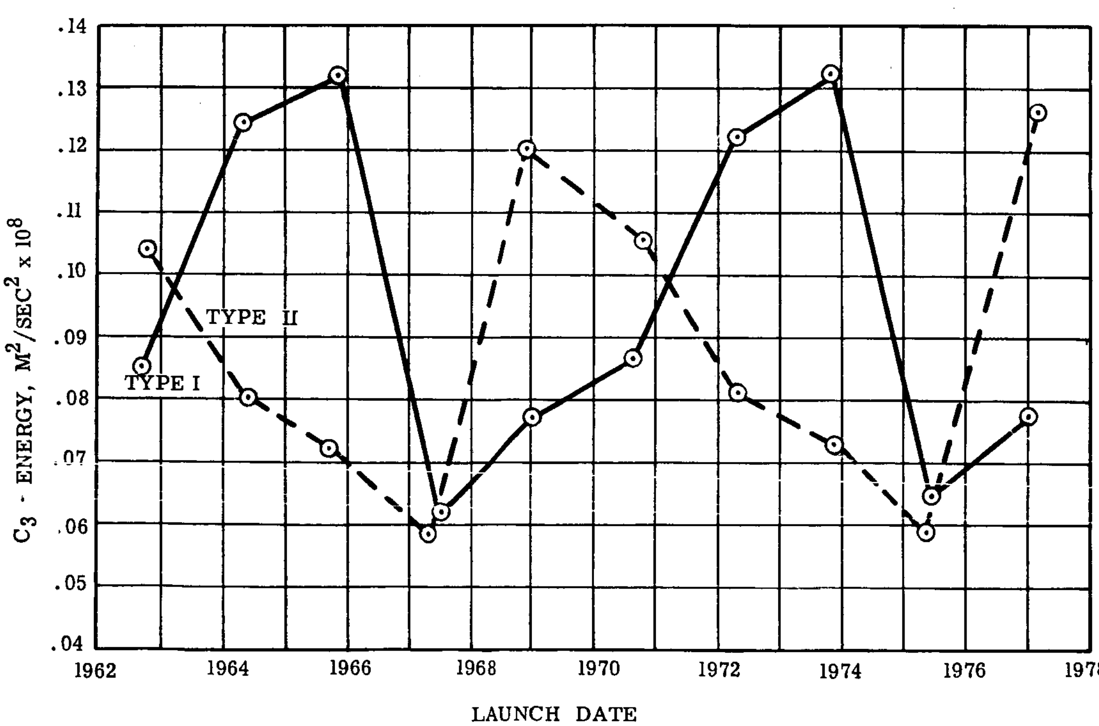


Figure 2.5-2. Absolute Minimum Energy for Venus

Orbiter and Lander weightwere considered from an all Lander to an all Orbiter system. In addition, it was necessary to treat the planetary orbit in a parametric fashion so that orbits from a circular one to one with a perifocus of 1000 n. mi. and an apofocus of 19, 000 n. mi. were considered.

Using the values of orbit insertion velocity requirements given in Figures 2.2-1 and 2.2-2 and orbit insertion propulsion capability given in Figure 2.4-1, it is possible to determine the allowable Lander weight as a function of launch data, trip time, size of Orbiter and planetary orbit. After choosing the particular trip time for a given launch date that yields a maximum payload weight (Orbiter plus Lander) the results were plotted for the various opportunities in Figures 2.5-3 through 2.5-14 assuming the Saturn C1B launch vehicle From these curves and assuming a 30 day launch window, the Orbiter weight as a function of Lander weight was obtained and is plotted in Figures 2.5-15 through 2.5-25. For the Venus 1967 opportunity, however, in order to obtain a 45 day launch window without a resulting serious degradation in payload capability a combination of a type I and type II trajectory was considered (Figures 2.5-3 and 2.5-4). During the first portion of the launch window the type II trajectory is chosen while the type I trajectory is used during the latter portion of the launch window. This approach is possible for the Venus 1967 opportunity since the type II trip times are not significantly greater than those obtained with the type I trajectories.

In order to maximize the Orbiter plus Lander weight, it was observed that for a number of opportunities it would be necessary to vary the total spacecraft weight within the period of the launch window. Since the Lander and Orbiter weight is held constant, this variation results from varying the fuel within the orbit insertion propulsion system. For instance, in the Mars 1969 opportunity the fuel in the orbit insertion propulsion system would have to be increased each day so as to achieve the same orbit with the same Orbiter weight. For other opportunities the weight may stay constant for a number of days and then decrease during the latter portion of the window. Using this technique to maximize total Orbiter plus Lander weight requires a capability to off-load or on-load fuel in the Voyager system throughout the launch window. However, the performance gain with this technique is large enough to warrant its consideration.

2.6 VOYAGER CONCEPTS

The next step in narrowing down the number of possible Voyager systems from the parametric results given in the previous section is to identify several of the most desirable systems for each opportunity. This was accomplished by identifying concepts that would maximize the value of certain scientific information while retaining (where possible) the same size modules (Orbiter and Lander) through the various opportunities.

An estimate was made early in the study of the Orbiter weight for several missions such as a Mars TV mapper, a Venus radar mapper (both utilized as a communications relay for the Lander(s)) and a minimum communications relay Orbiter. Having narrowed down the possible Orbiter weights, several concepts were chosen: one favoring the landing mission (large Lander(s) and eccentric orbit) and the other favoring the Orbiter mapping mission (smaller Lander(s) and circular orbit).

Another variable that was considered was the time for Lander release: before the Orbiter is injected into orbit (direct entry) or after orbit insertion. The first choice is by far the most desirable from a weight consideration, but the second choice may be desirable from the standpoint of (1) achieving a reduced entry environment (lower entry velocity), (2) more effective utilization of the Lander since it could be used to explore interesting areas that were discovered by the Orbiter TV system, (3) obtaining a tighter control over the entry corridor or (4) placing the Lander in an area that may not be accessible with a direct entry.

Considering these various system possibilities, a number of Voyager concepts were identified early in the study for more detailed analysis. These concepts for various opportunities are given in Tables 2.6-1 to 2.6-10. (In a number of cases, the values given in these tables may not check exactly with the values given in Section 2.5 since these concepts take into account the fuel that may be required for Orbiter retardation, prior to orbit insertion, in order to maintain adequate line of sight with the Lander prior to impact. See Volume III, Section 4.0.)

Considering these system concepts it is now possible to identify certain interplanetary trajectory characteristics such as launch windows, trip times, arrival geometry, etc. These characteristics are given in Table 2.6-11.

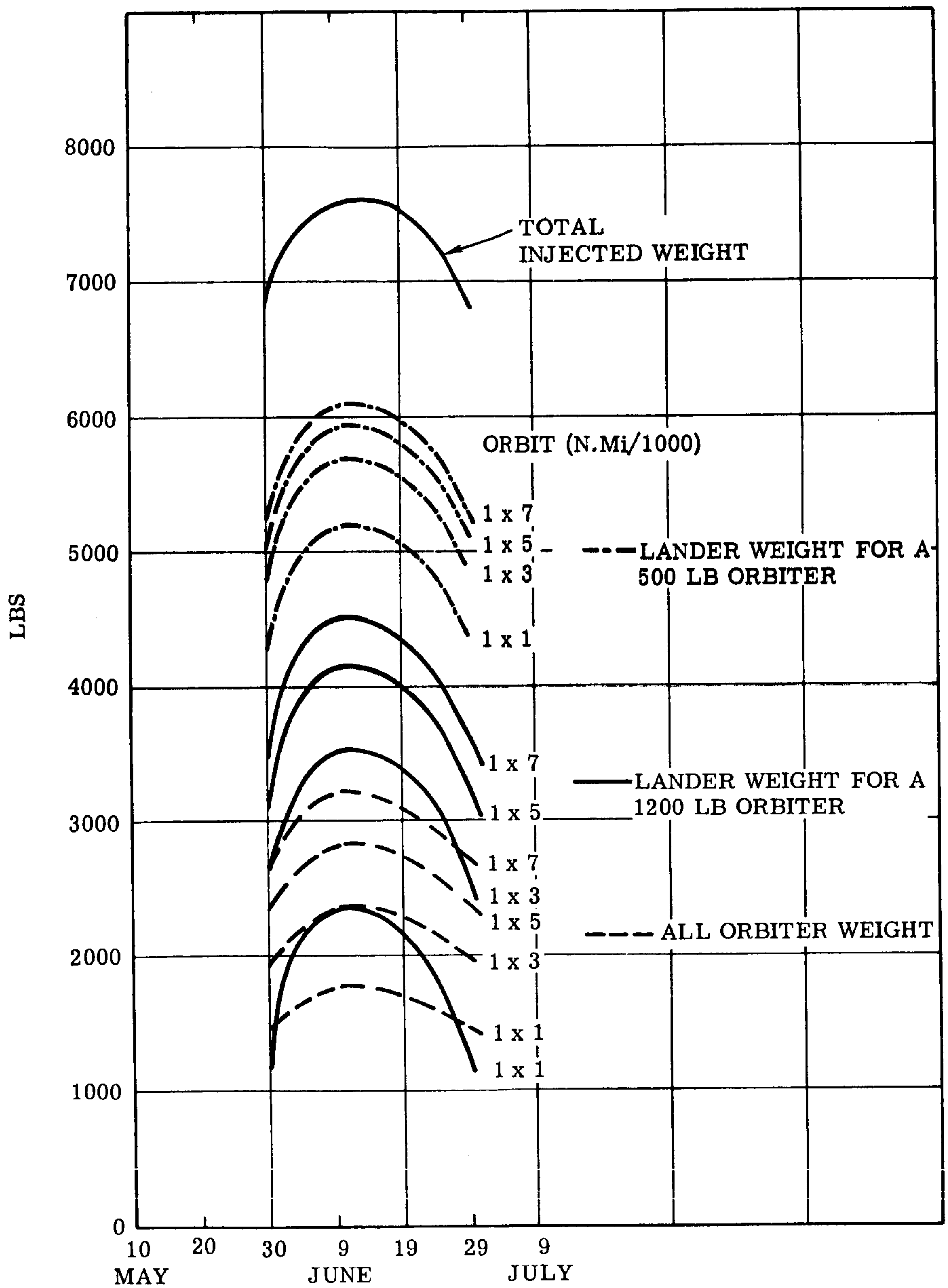


Figure 2.5-3. Launch Vehicle Capability, Venus 1967 (Type I Trajectory)

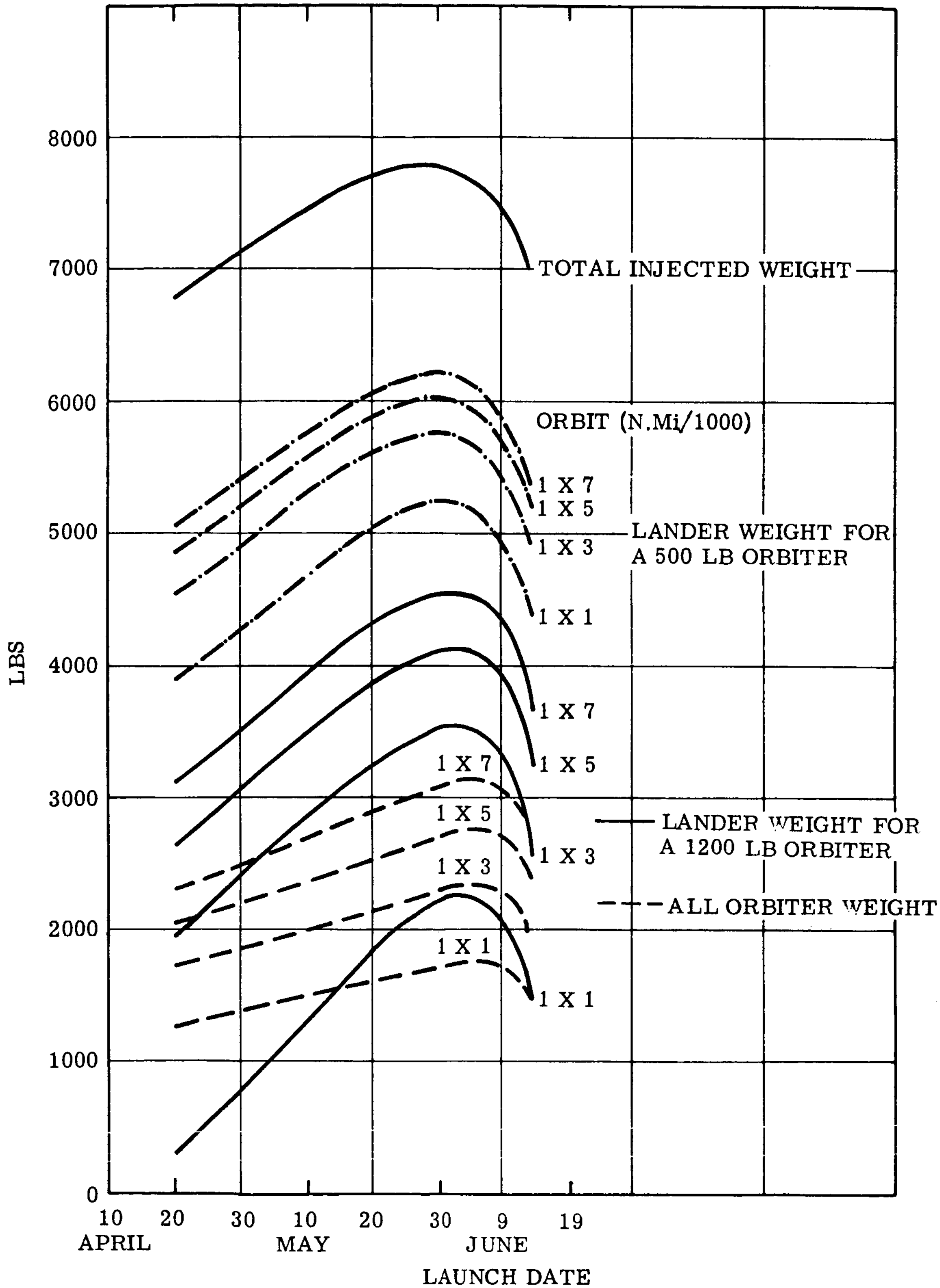


Figure 2.5-4. Launch Vehicle Capability, Venus 1967 (Type II Trajectory)

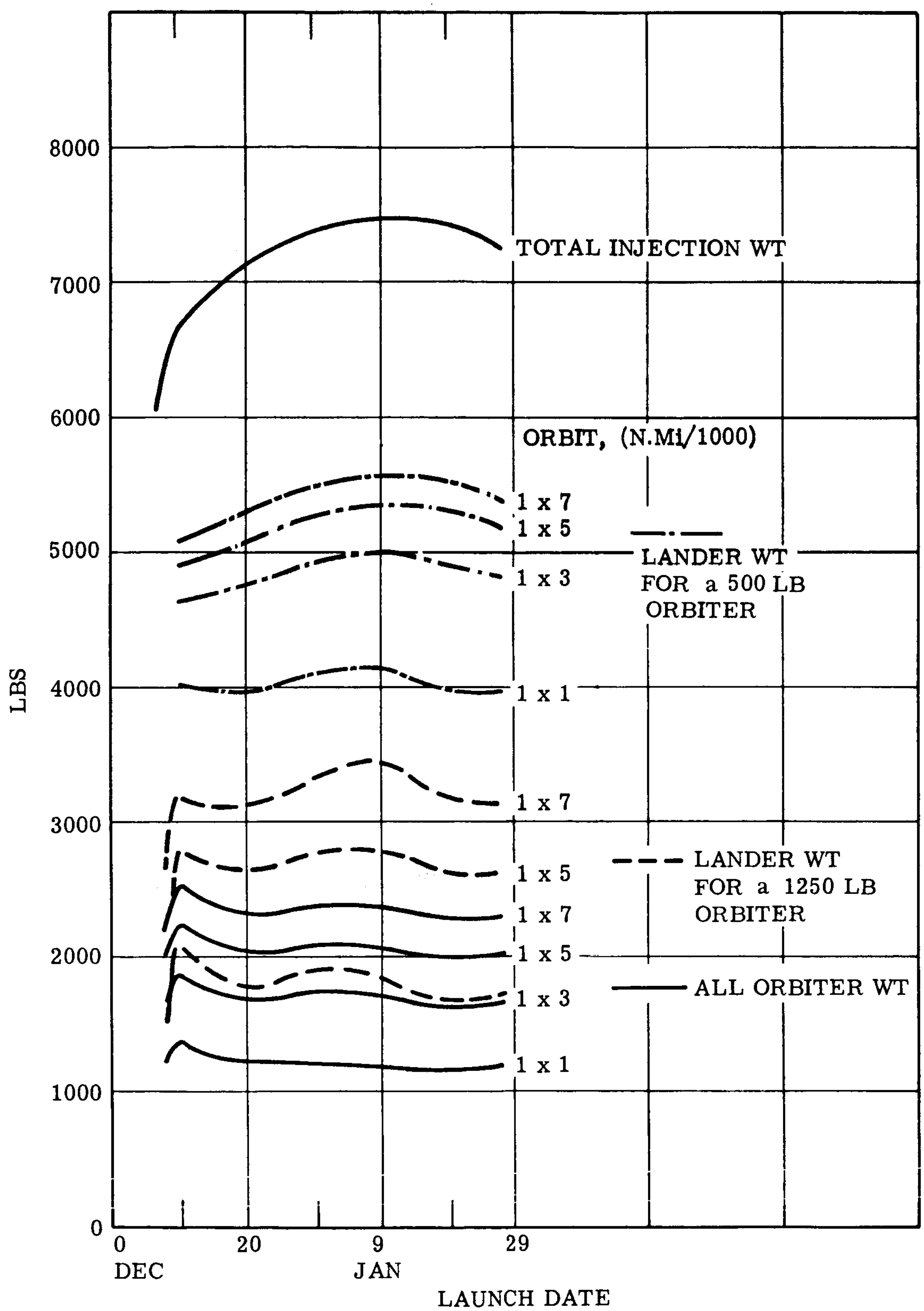


Figure 2.5-5. Launch Vehicle Capability, Venus 1968-1969 (Type I Trajectory)

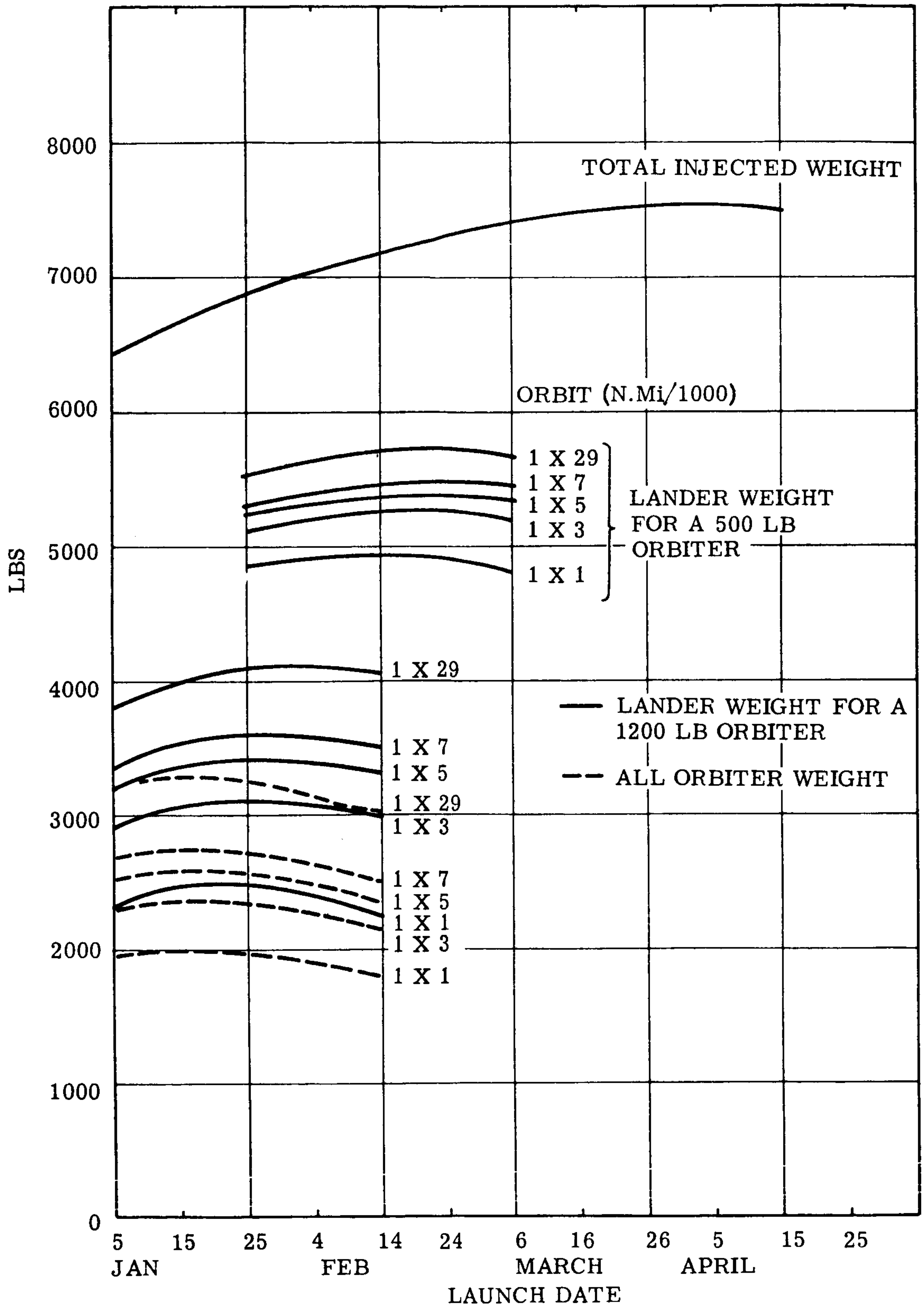


Figure 2.5-6. Launch Vehicle Capability, Mars 1969 (Type II Trajectory)

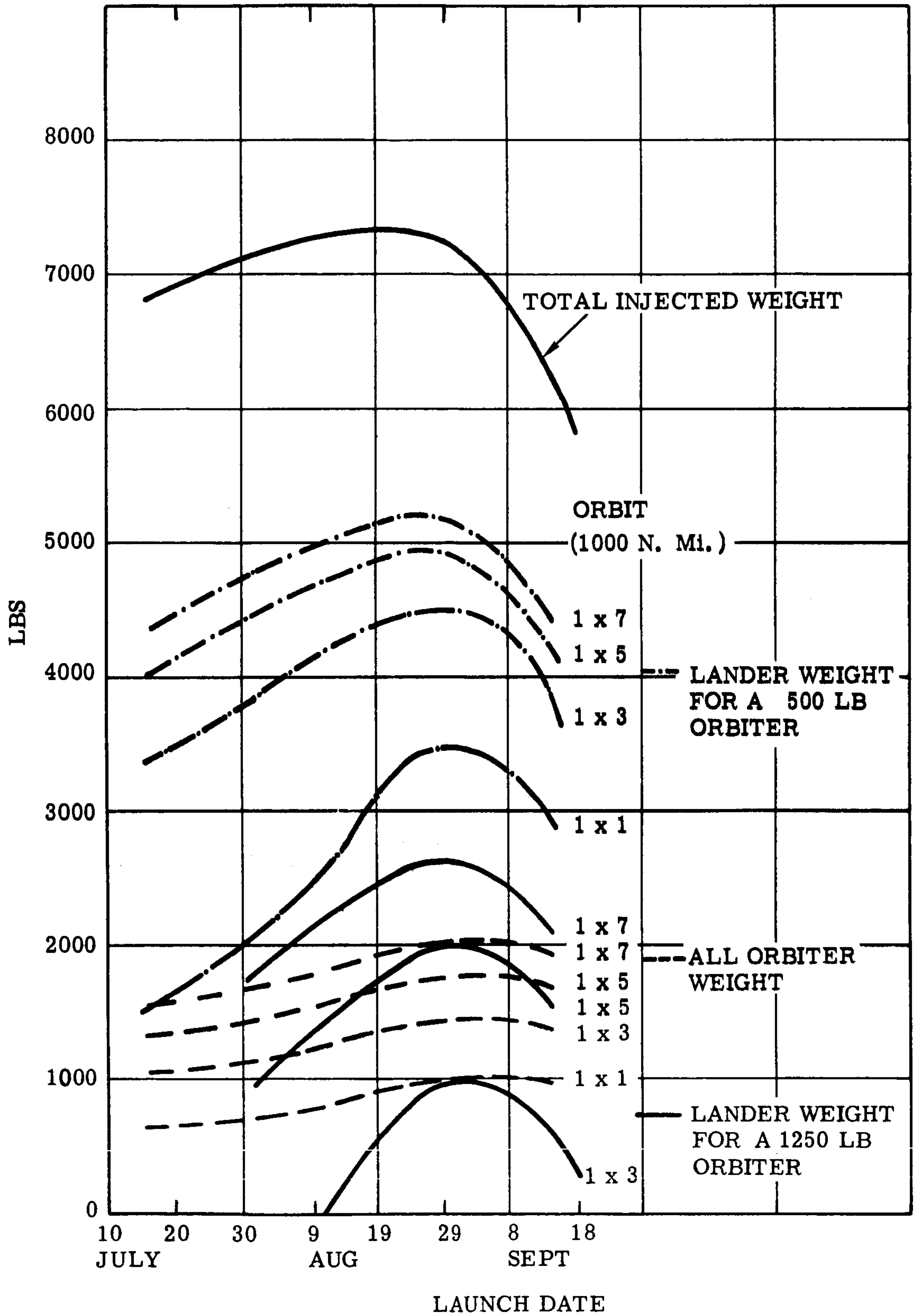


Figure 2.5-7. Launch Vehicle Capability, Venus 1970 (Type I Trajectory)

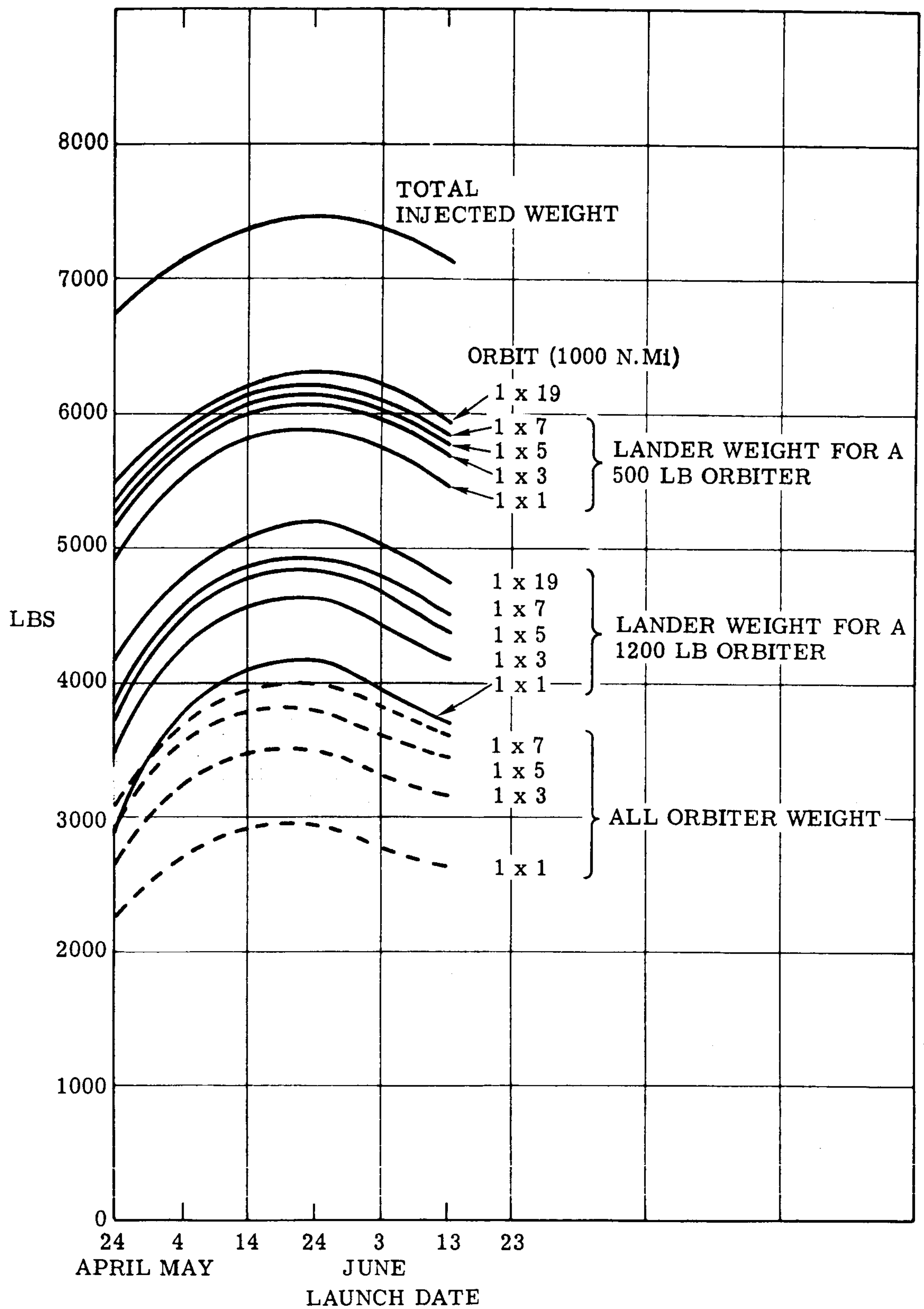


Figure 2.5-8. Launch Vehicle Capability, Mars 1971 (Type I Trajectory)

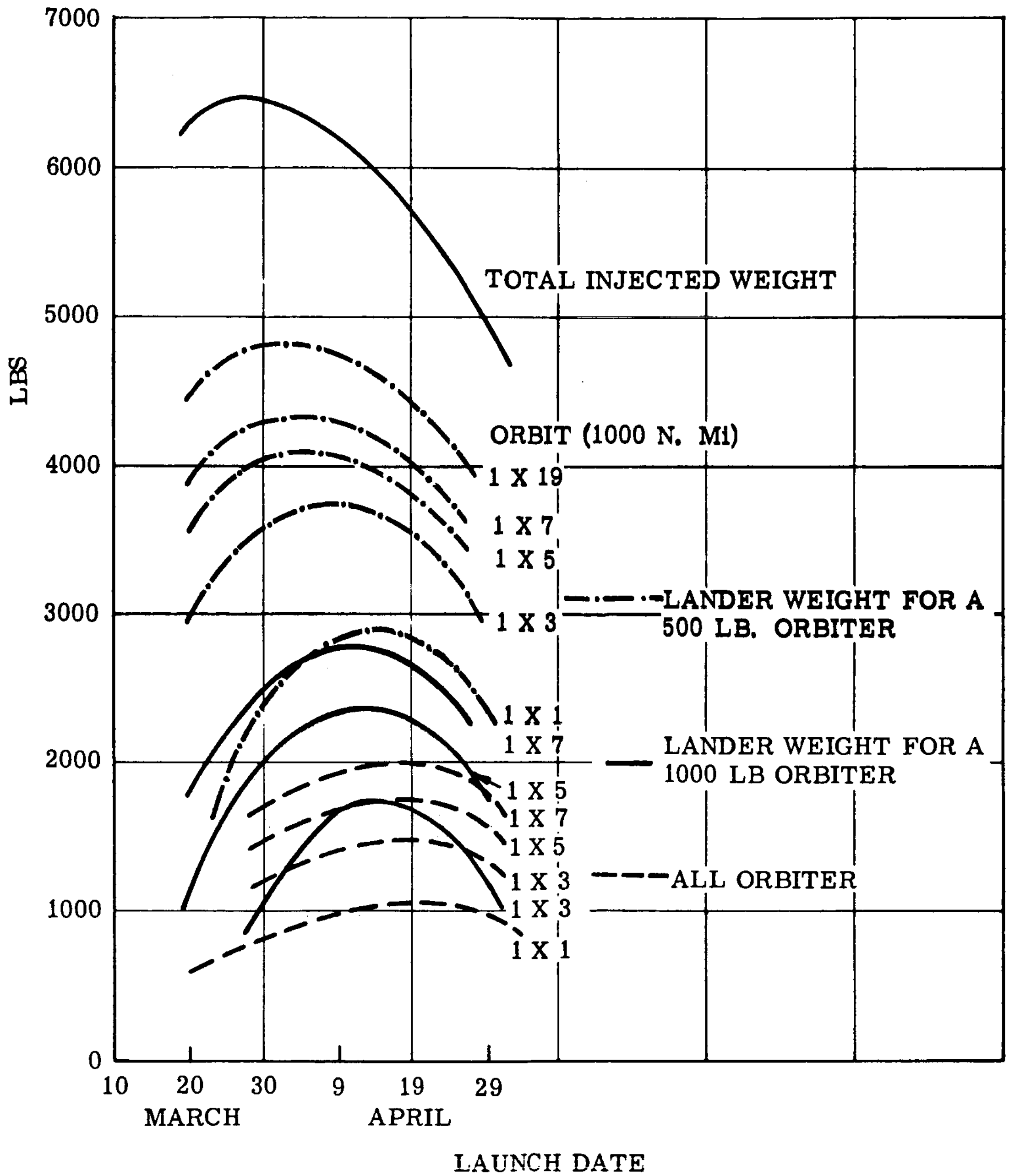


Figure 2.5-9. Launch Vehicle Capability, Venus 1972 (Type I Trajectory)

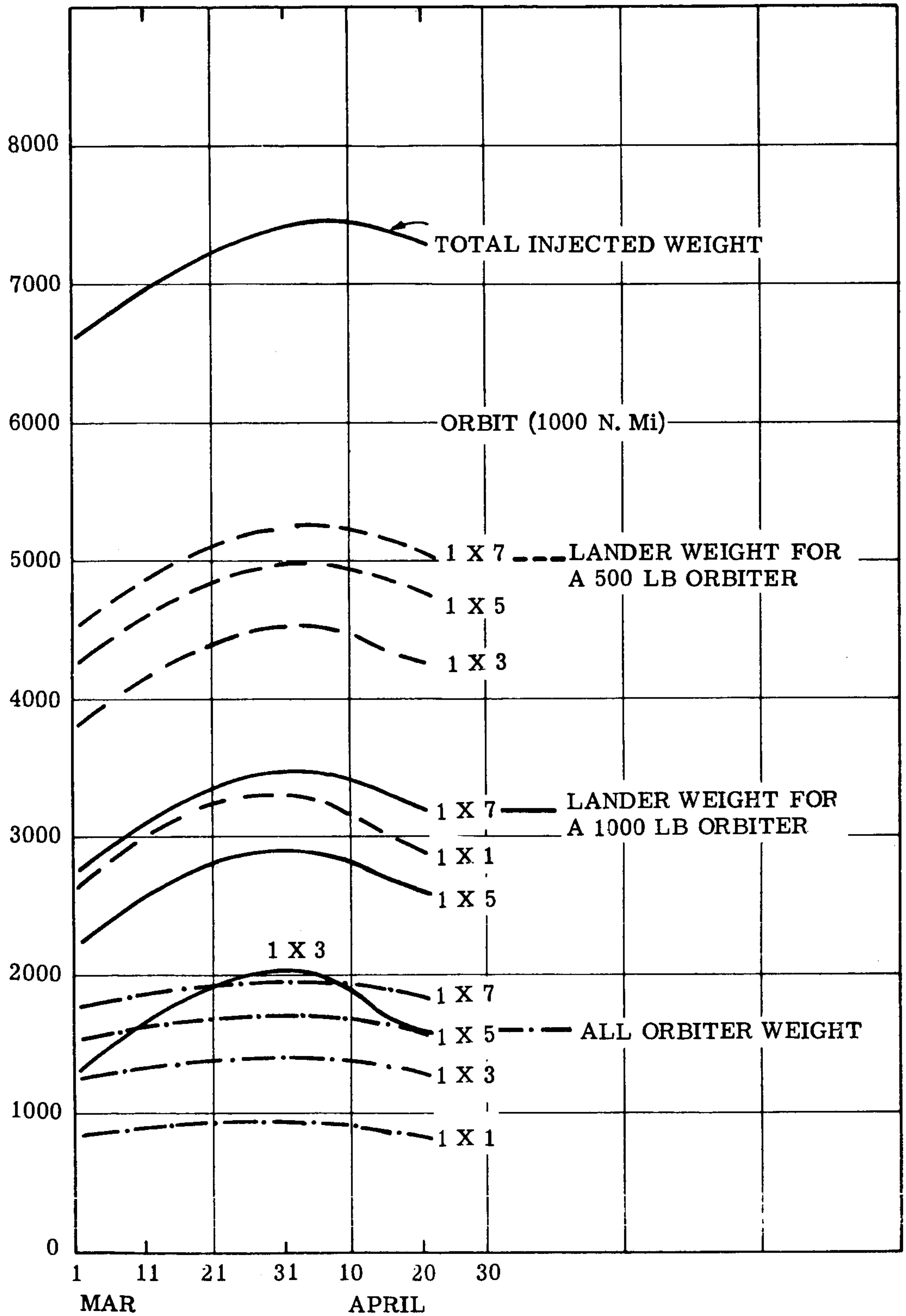


Figure 2.5-10. Launch Vehicle Capability, Venus 1972 (Type II Trajectory)

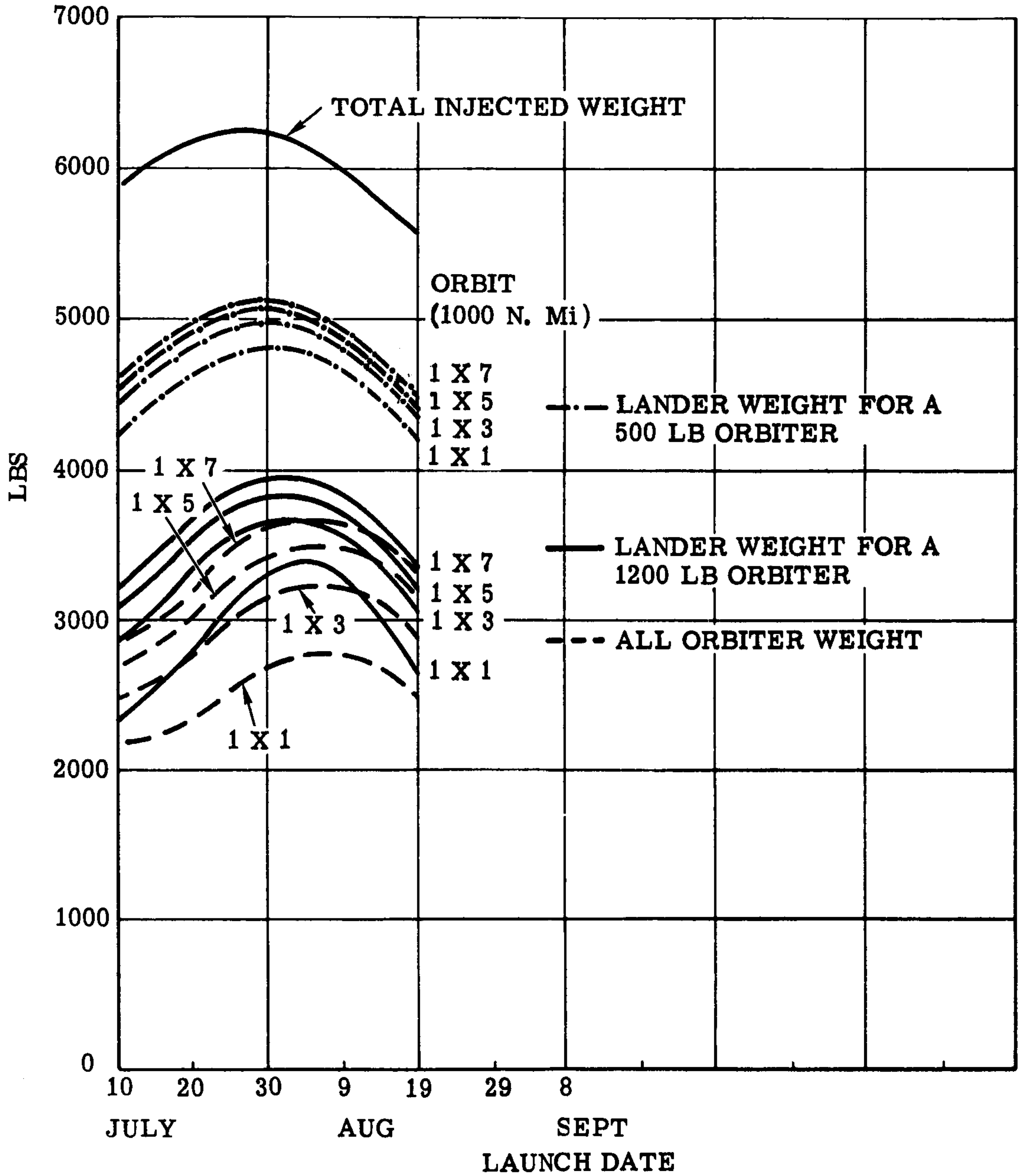


Figure 2.5-11. Launch Vehicle Capability, Mars 1973 (Type I Trajectory)

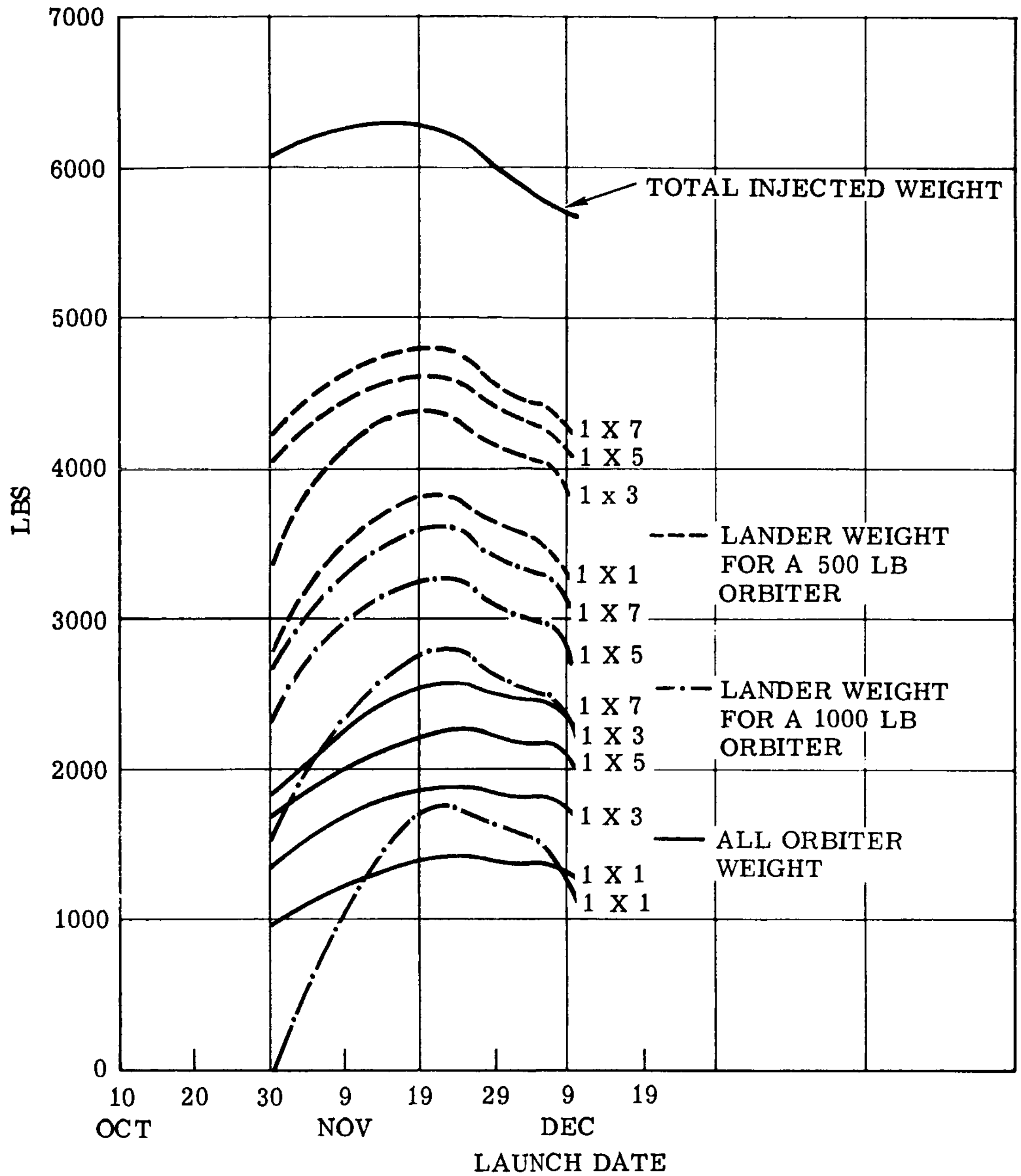


Figure 2.5-12. Launch Vehicle Capability, Venus 1973 (Type I Trajectory)

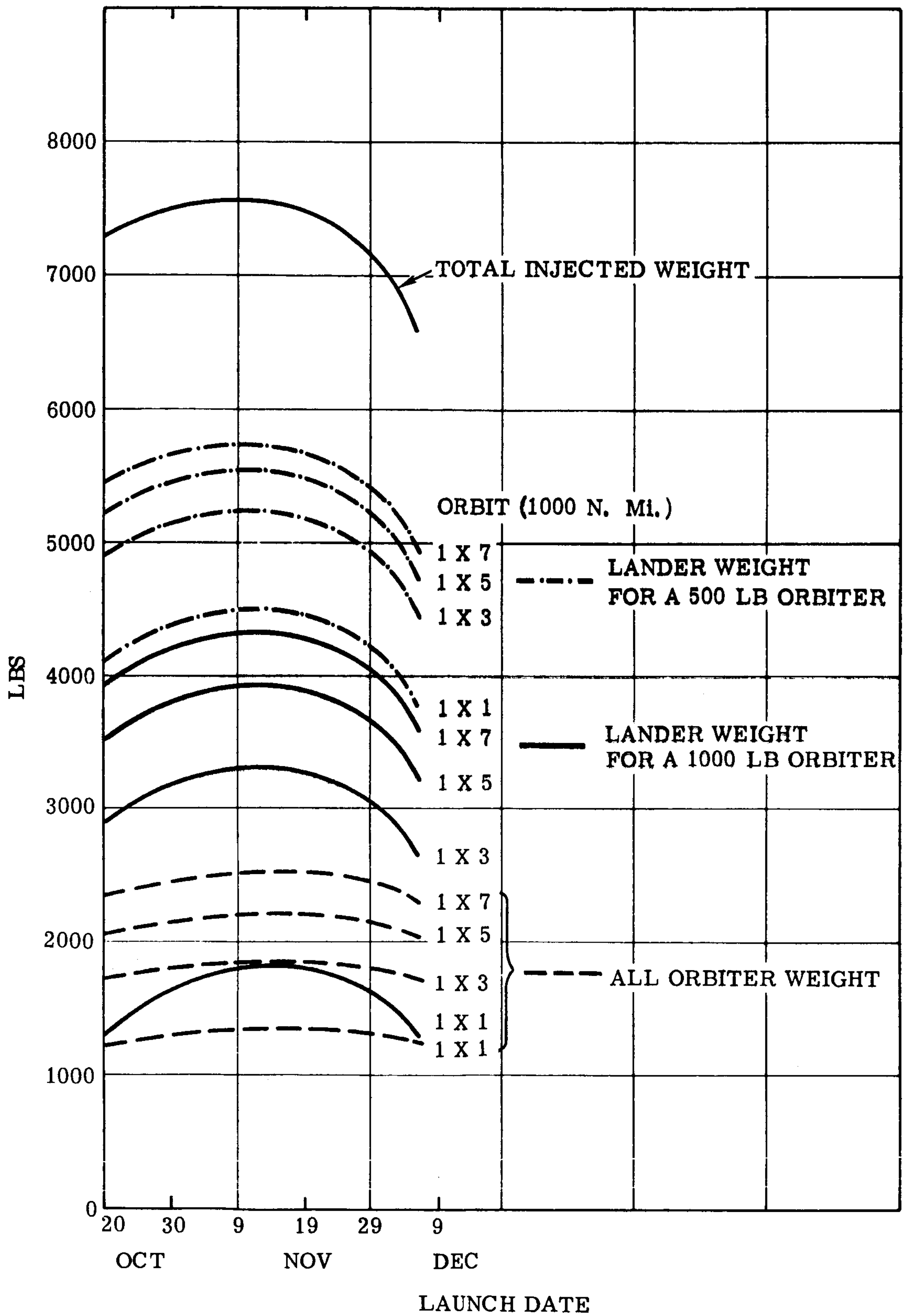


Figure 2.5-13. Launch Vehicle Capability, Venus 1973 (Type II Trajectory)

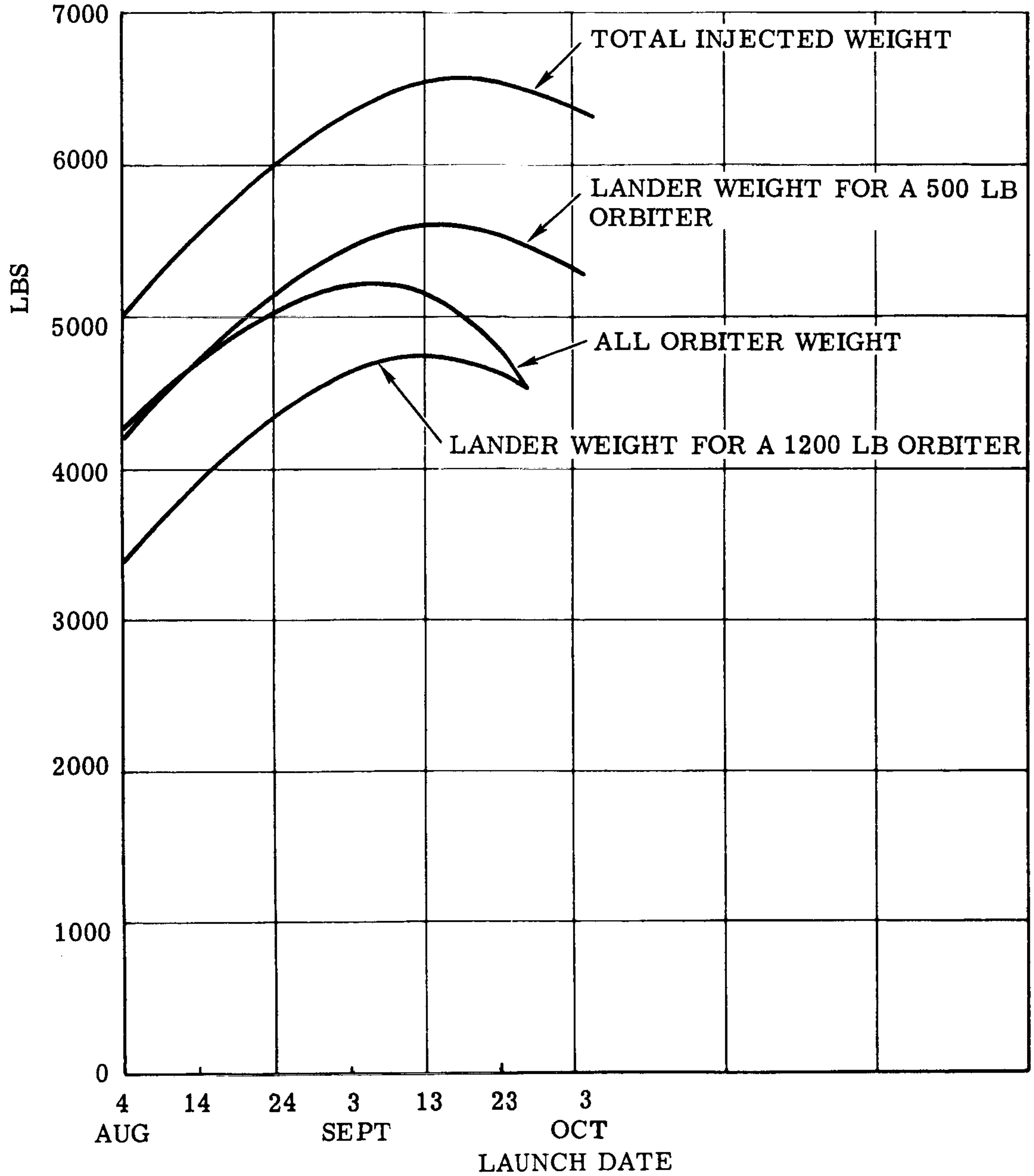


Figure 2.5-14. Launch Vehicle Capability, Mars 1975 (Type II Trajectory)

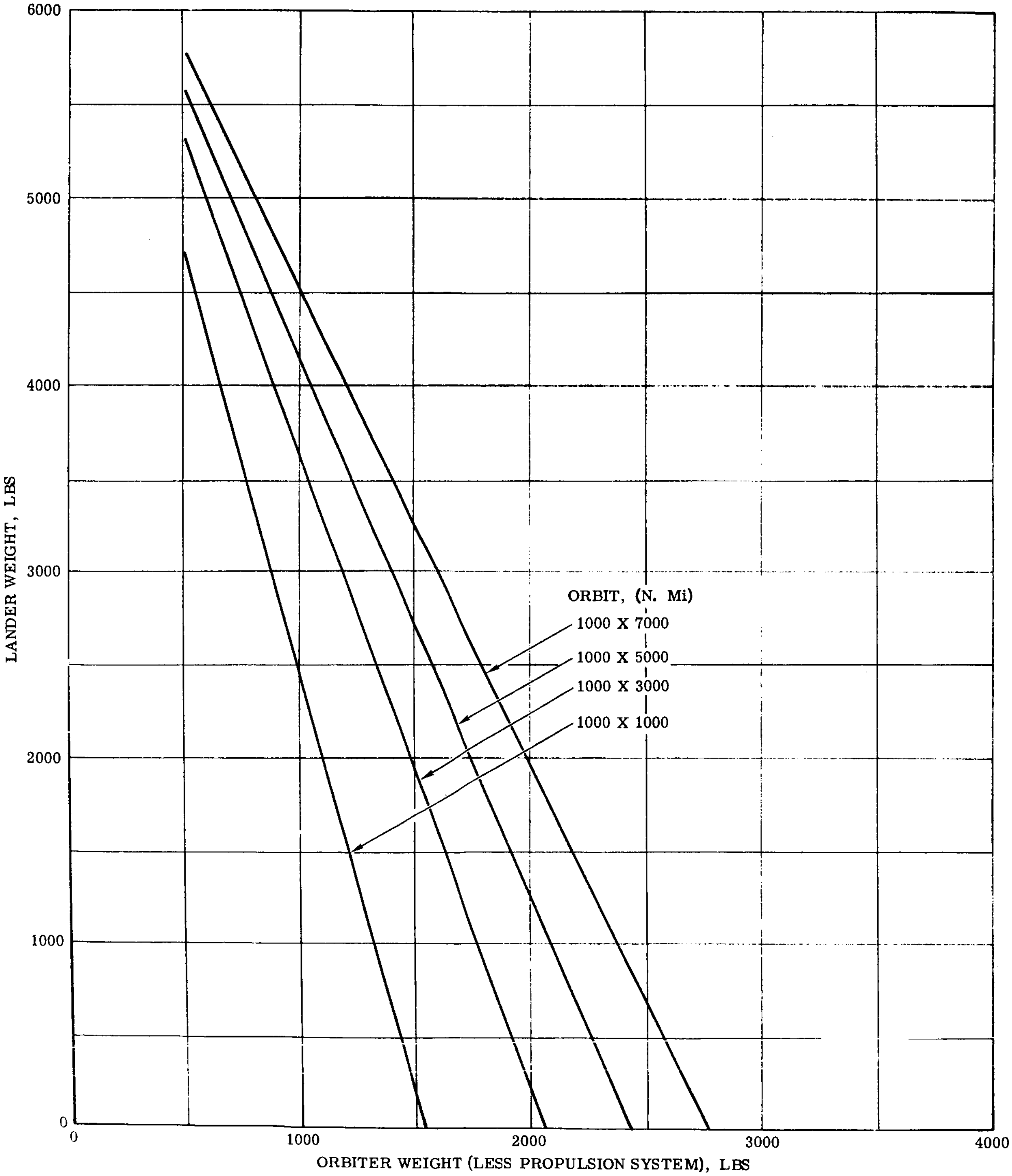


Figure 2.5-15. Orbiter vs. Lander Weight, Venus 1967 (45 Day Launch Window)

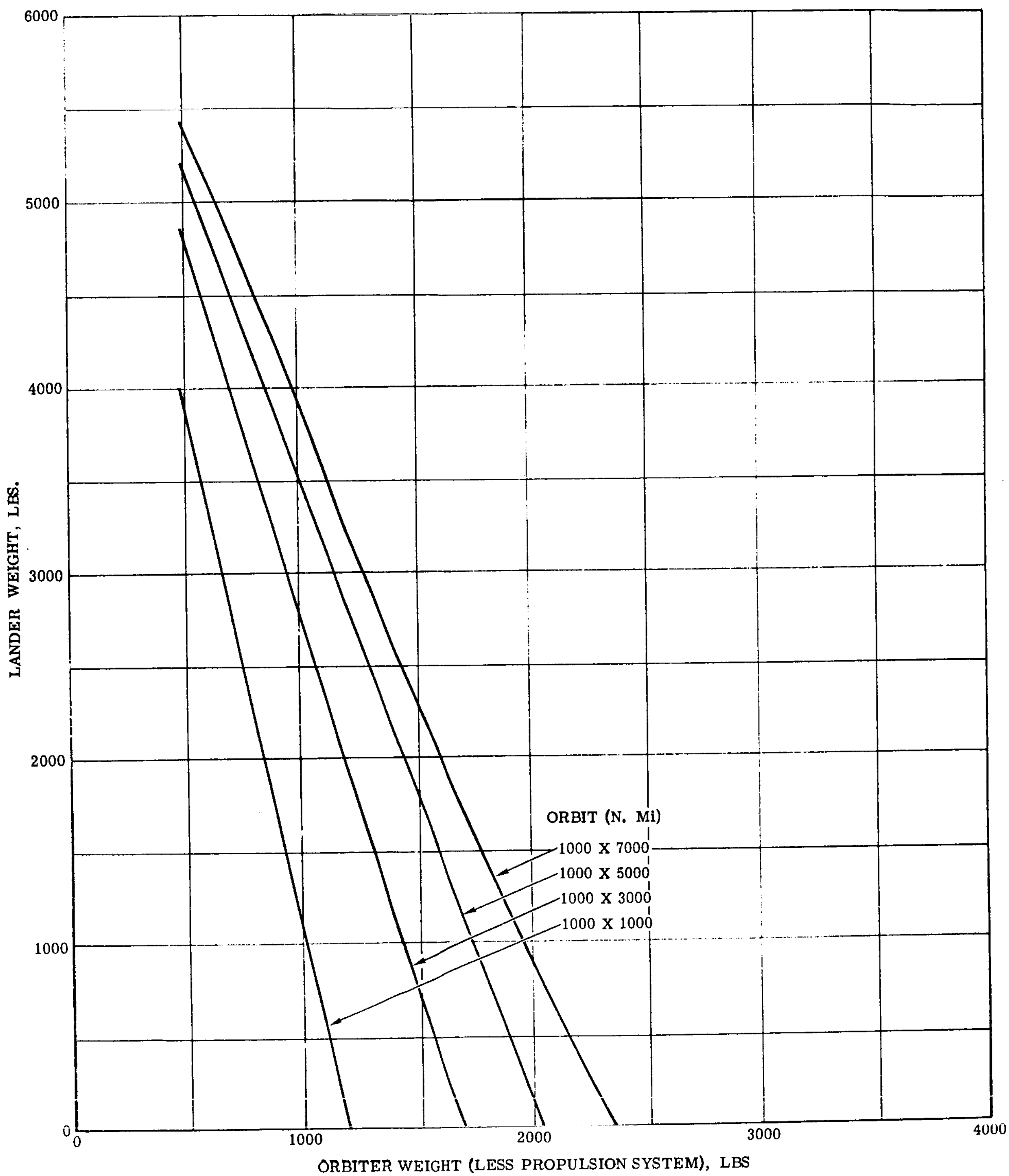


Figure 2.5-16. Orbiter vs. Lander Weight, Venus 1968 (30 Day Launch Window)

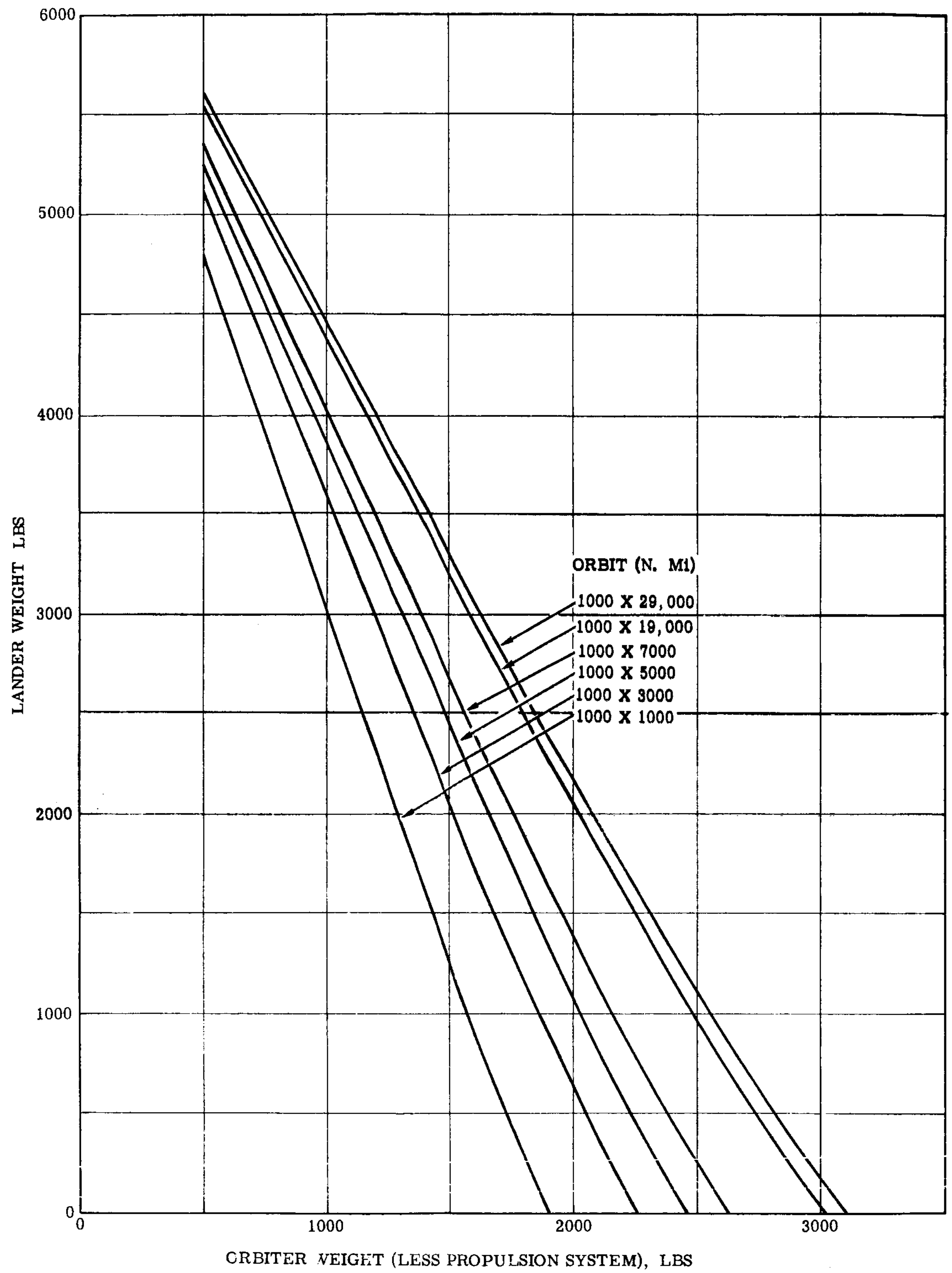


Figure 2.5-17. Orbiter vs. Lander Weight, Mars 1969 (30 Day Launch Window)

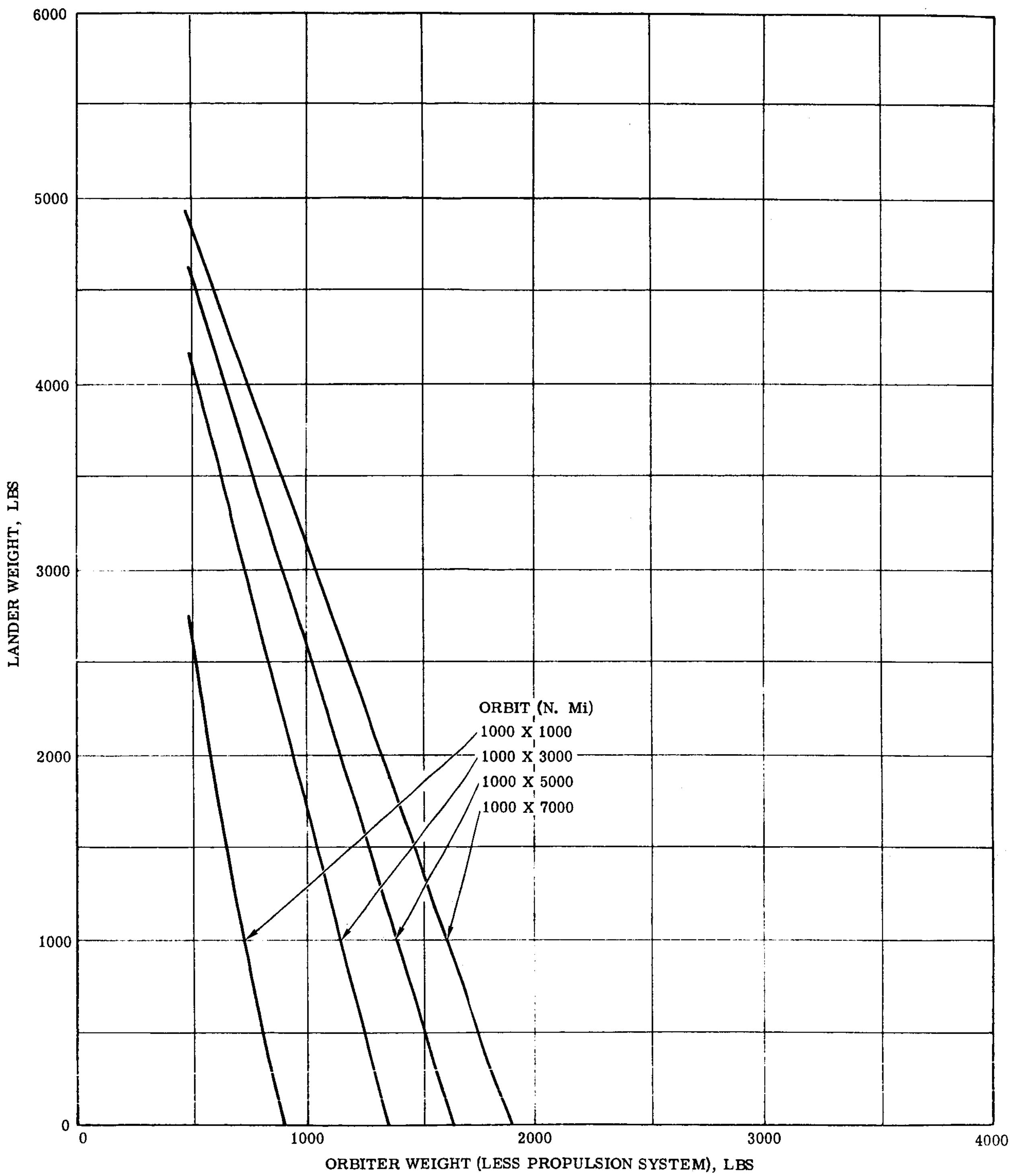


Figure 2.5-18. Orbiter vs. Lander Weight, Venus 1970 (30 Day Launch Window)

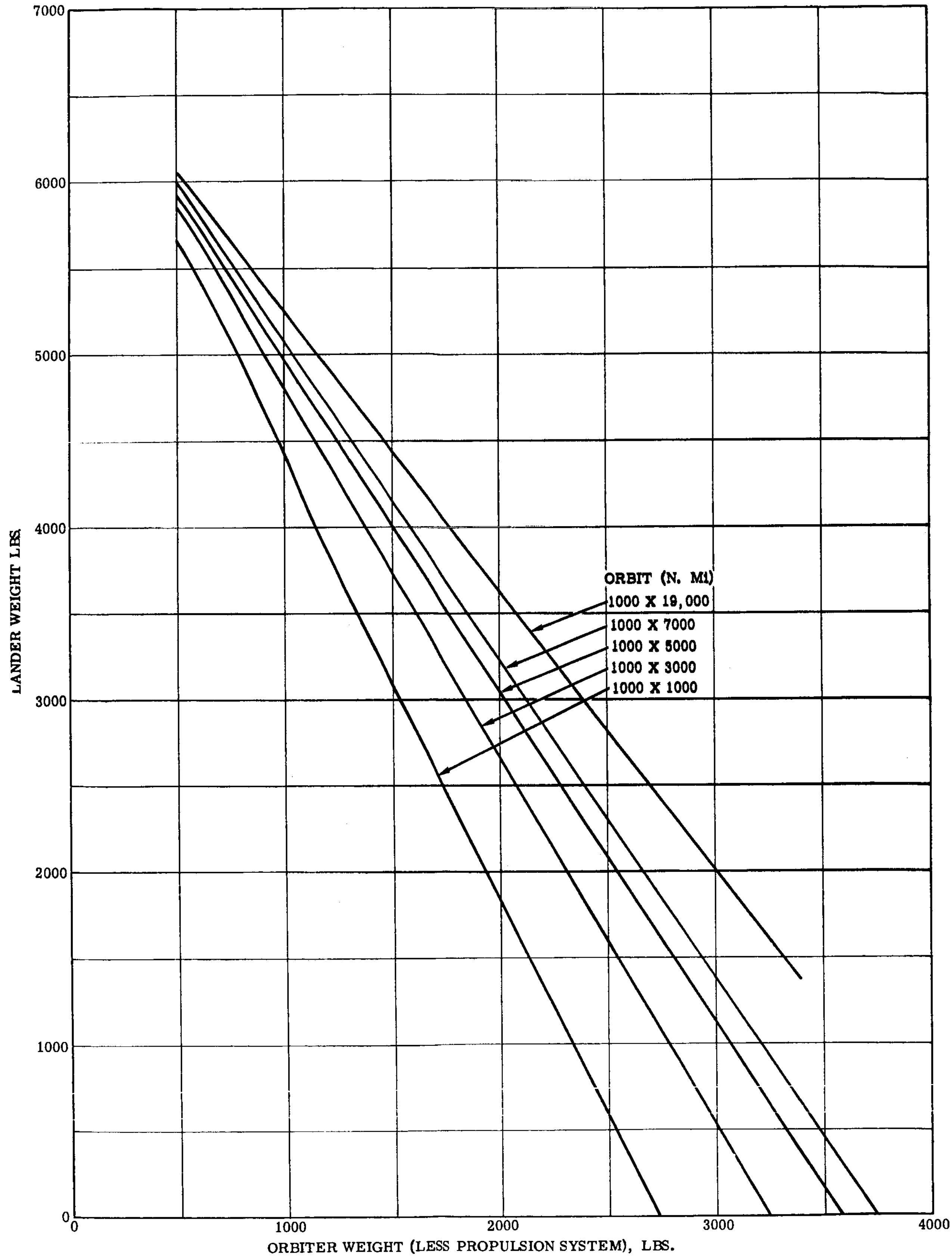


Figure 2.5-19. Orbiter vs. Lander Weight, Mars 1971 (30 Day Launch Window)

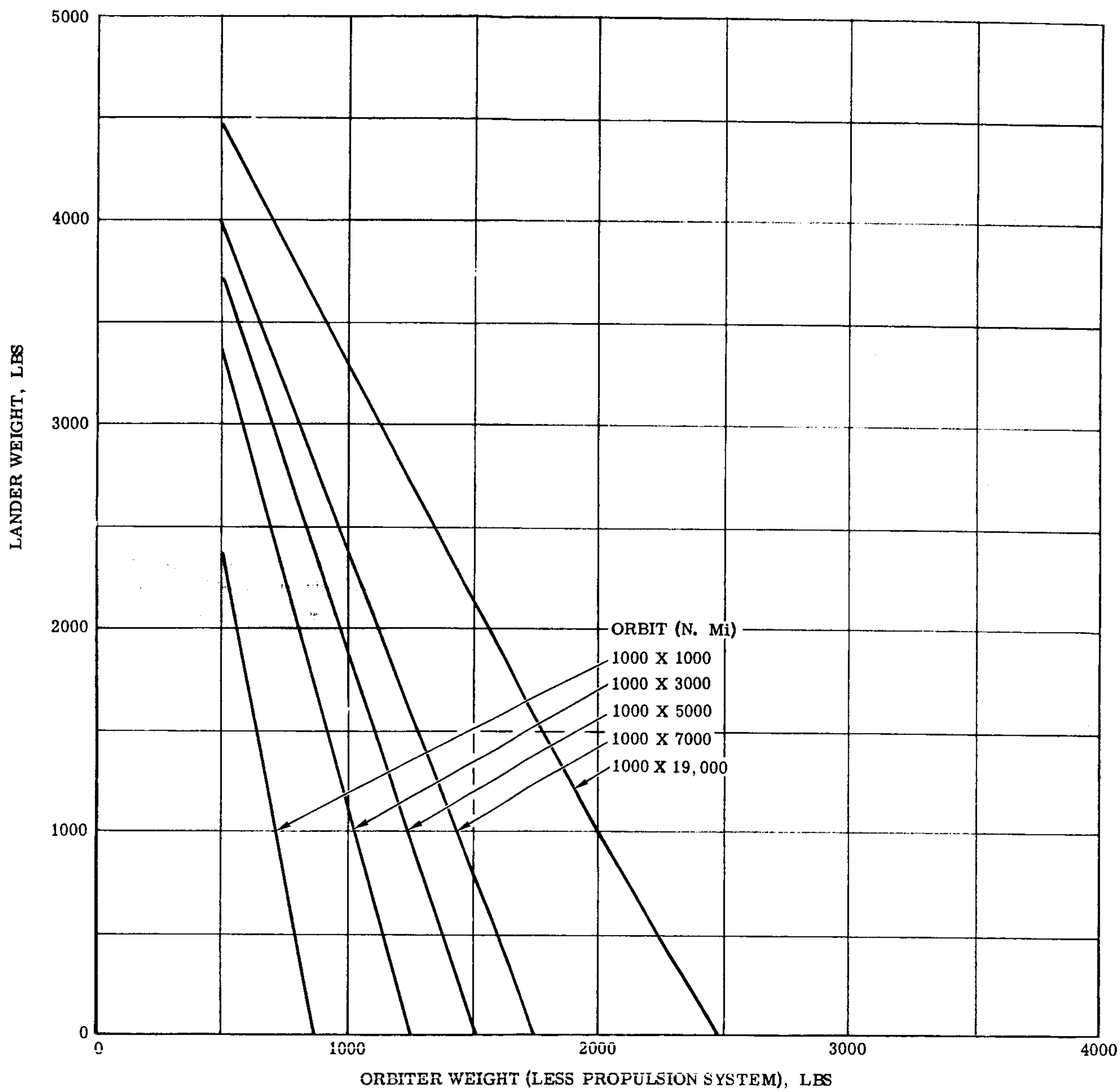


Figure 2.5-20. Orbiter vs. Lander Weight, Venus 1972, Type I (30 Day Launch Window)

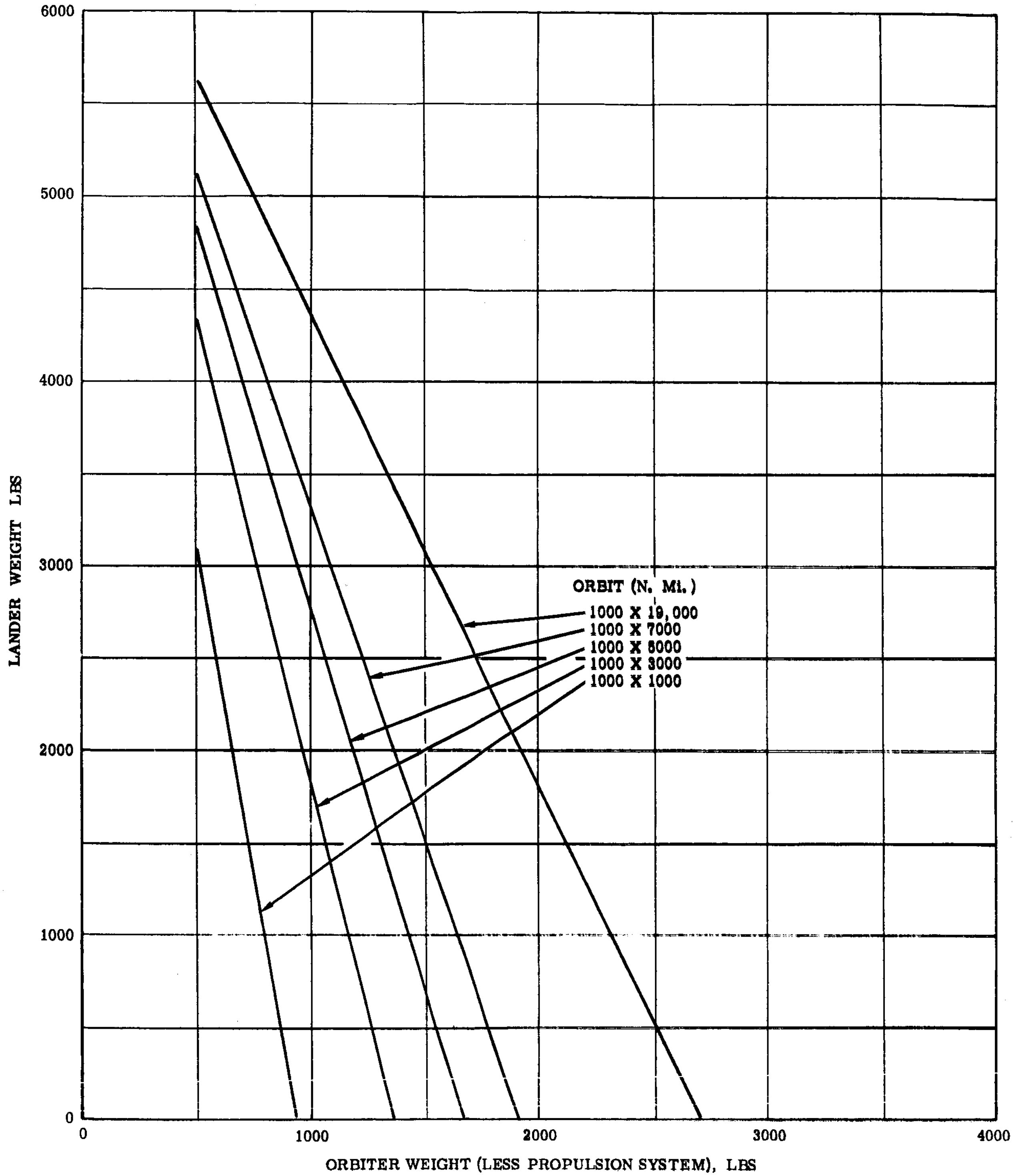


Figure 2.5-21. Orbiter vs. Lander Weight, Venus 1972, Type II (30 Day Launch Window)

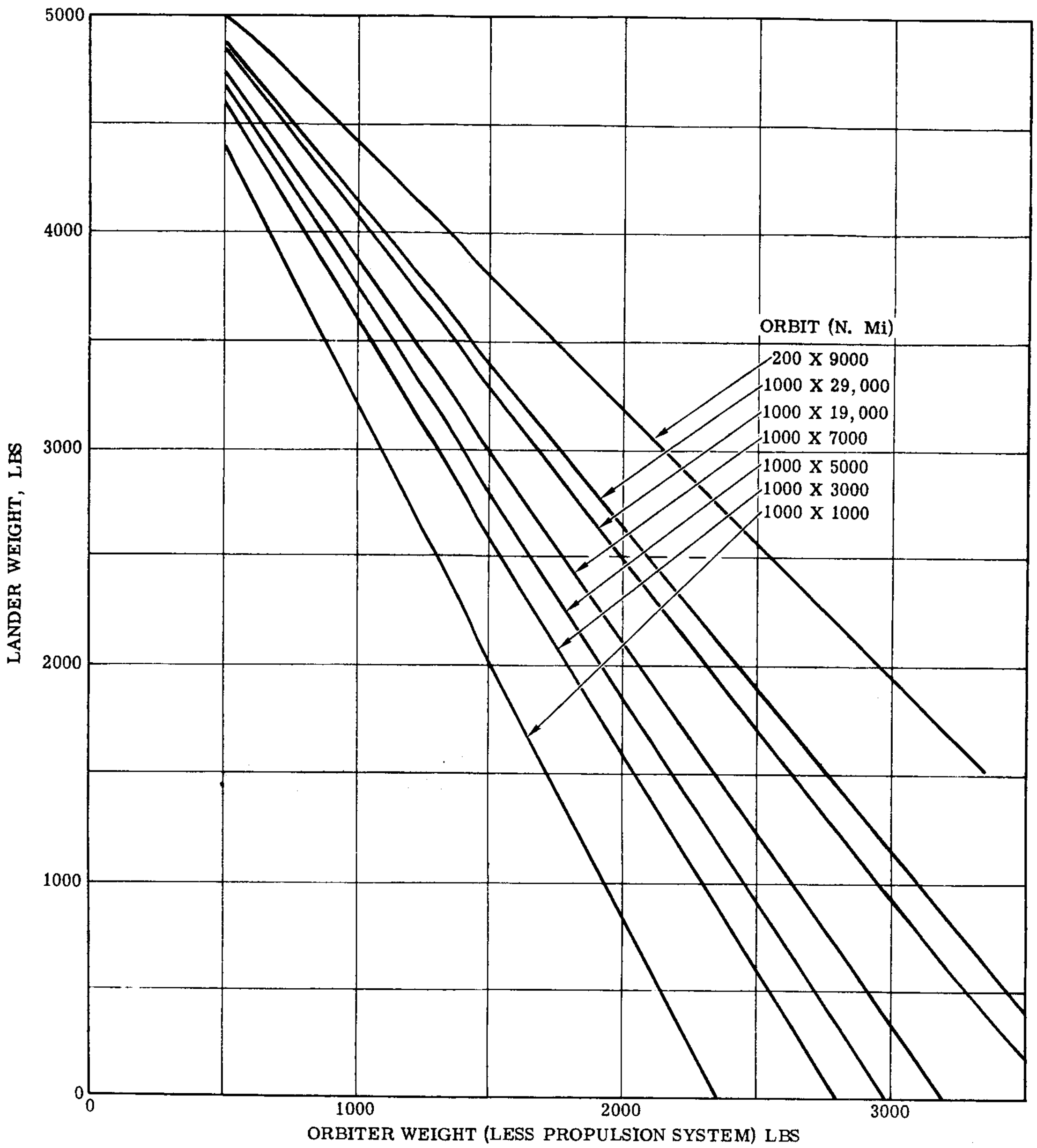


Figure 2.5-22. Orbiter vs. Lander Weight, Mars 1973, Type I (30 Day Launch Window)

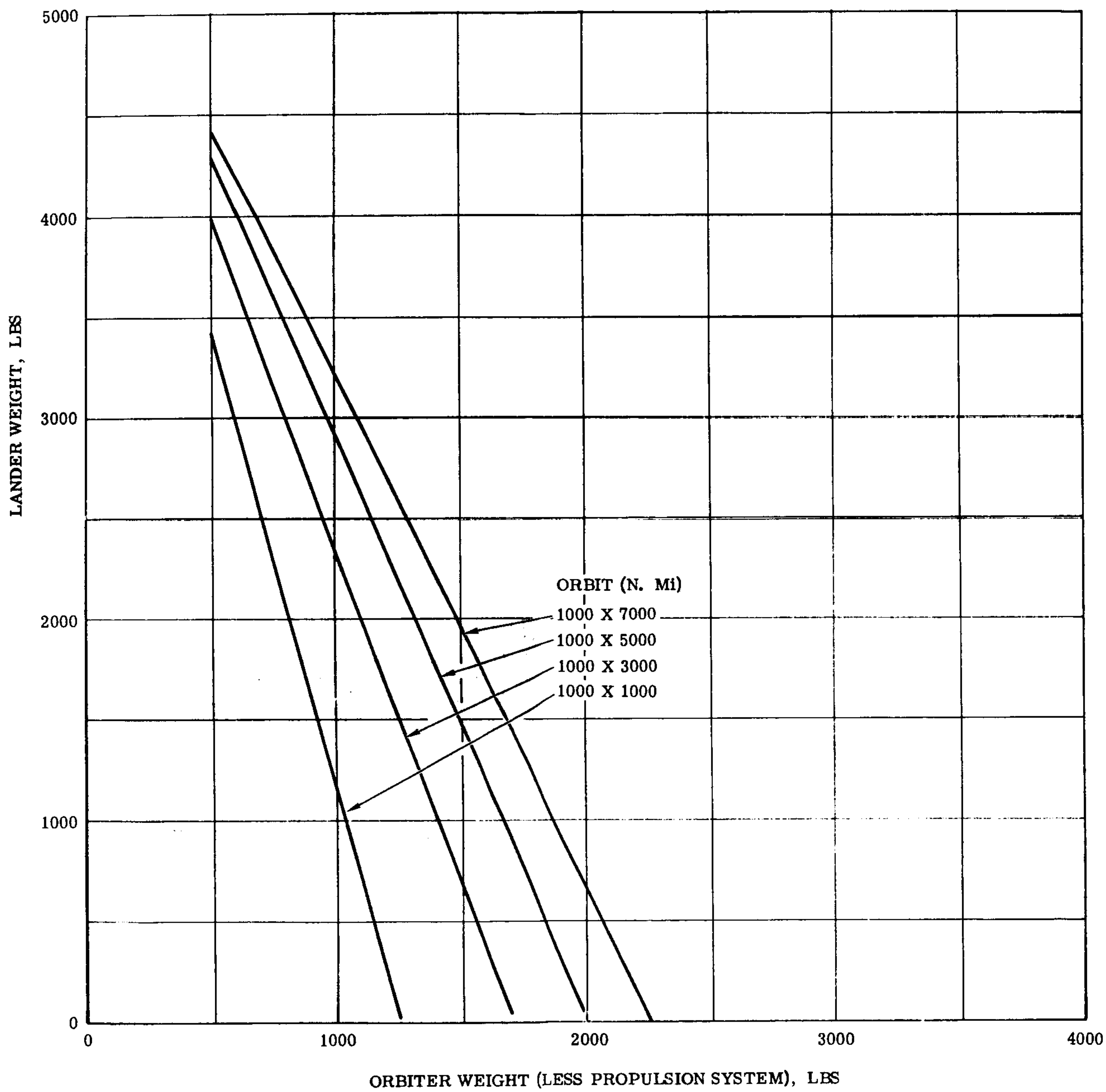


Figure 2.5-23. Orbiter vs. Lander Weight, Venus 1973, Type I (30 Day Launch Window)

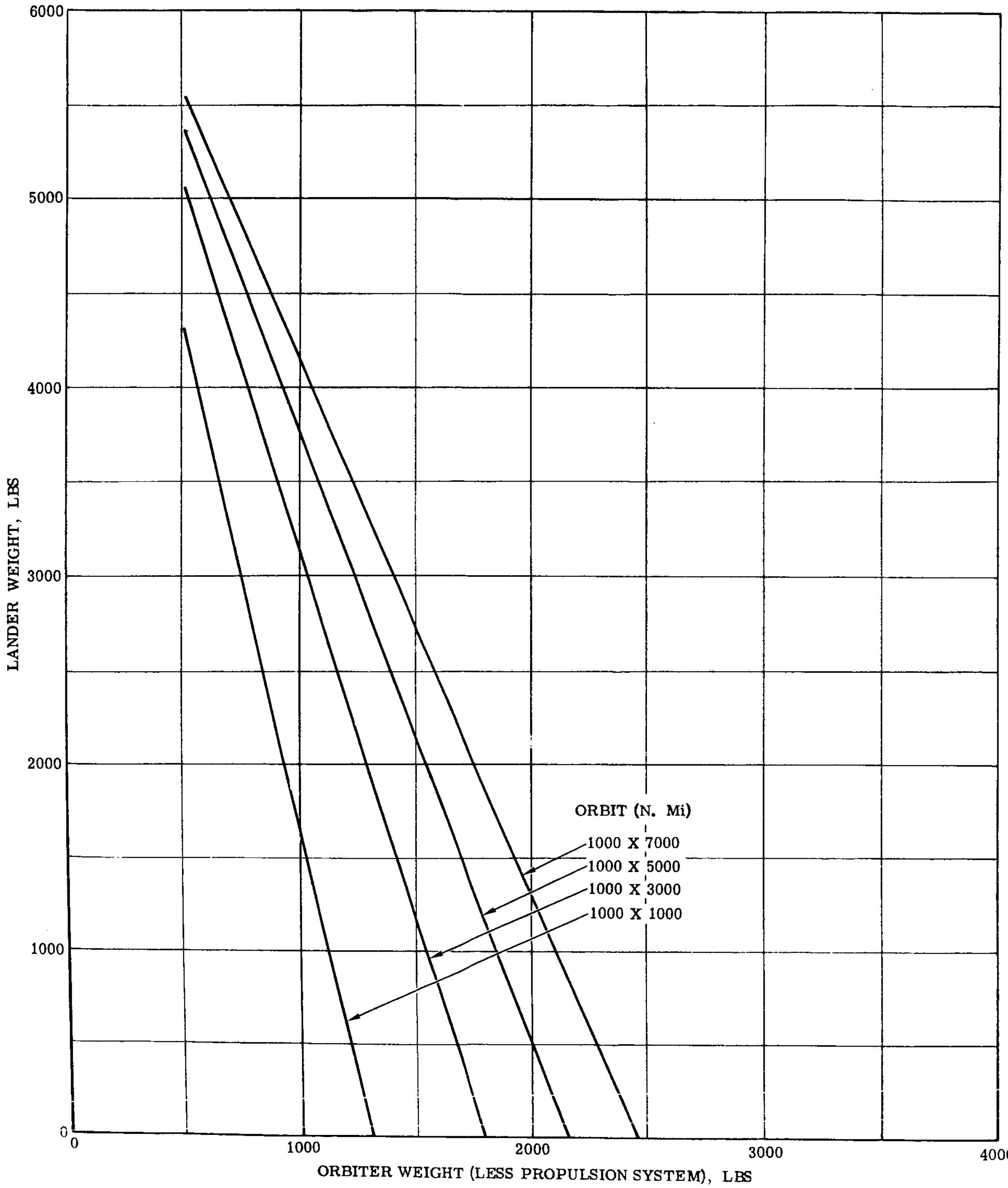


Figure 2.5-24. Orbiter vs. Launch Weight, Venus 1973, Type II (30 Day Launch Window)

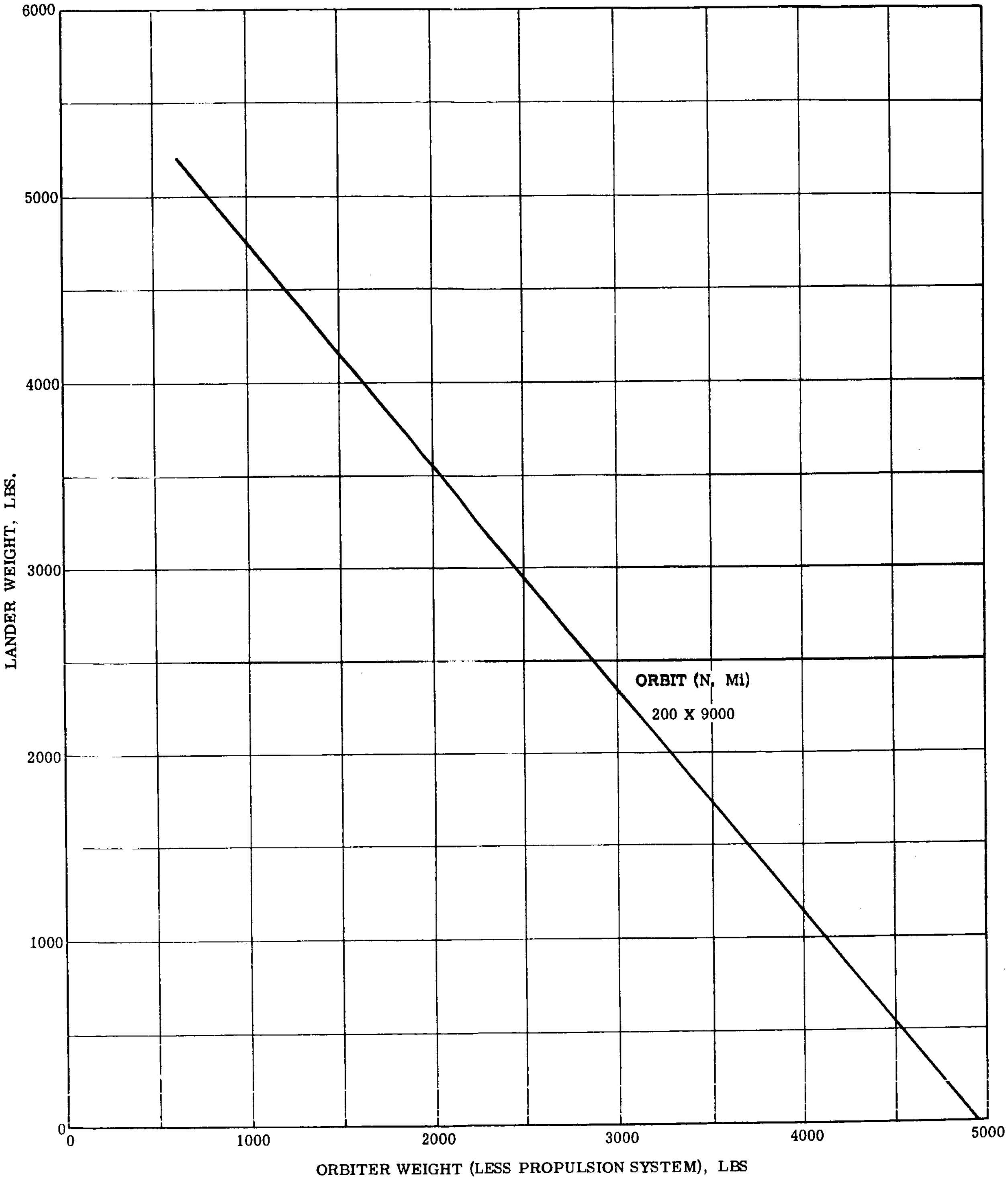


Figure 2.5-25. Orbiter vs. Launch Weight, Mars 1975, Type II (30 Day Launch Window)

TABLE 2.6-1. VENUS 1967 VOYAGER CONCEPTS

Description	67-1 Direct Entry TV Orbiter	67-2A Direct Entry Radar Orbiter	67-2B Direct Entry Radar Orbiter	67-3 One Entry Vehicle Direct One Entry Vehicle Released From Orbit Radar Orbiter
$W_{injected}$ (max.)	7500 lbs	7500 lbs	7500 lbs	7500 lbs
$W_{arrival}$	7125	7125	7125	7125
$W_{Orbiter}$ (less propulsion)	1165	1265	1265	1265
$W_{propulsion\ system}$ (orbit injection fuel and hardware)	3110	4388	3860	4360
$W_{Lander(s)}$ (less propulsion)	2550	716 716	975 975	520 820 *
$W_{Lander\ propulsion}$	300	20 20	25 25	35 125
Orbit (x 1000 n. mi.)	5 x 5	1 x 1	1 x 1 to 1 x 2.1	1 x 5
Surface Life Required	6 hrs	10 min 10 min	3 hrs 3 hrs	10 min 4 hrs

*Released at apofocus to achieve $\gamma = 7.5^{o}$ at 600,000 ft and $V_e = 28,000$ ft/sec

TABLE 2. 6-2. VENUS 1968-1969 VOYAGER CONCEPTS

Description	68-1 Direct Entry TV Orbiter	68-2A Direct Entry Radar Orbiter			68-2B Direct Entry Radar Orbiter		68-3 One Entry Vehicle Direct One Entry Vehicle Released From Orbit Radar Orbiter
$W_{injected}$ (max.)	7440		7440		7440		7440
$W_{arrival}$	7070		7070		7070		7050
$W_{Orbiter}$ (less propulsion)	1165		1265		1265		1265
$W_{propulsion\ system}$ (orbit injection fuel and hardware)	3055	(1) 5805	(2) 5070	(3) 4335	(1) 4805	(2) 3805	4285
$W_{Lander(s)}$ (less propulsion)	2550	0	715	715 715	975	975 975	520 820*
$W_{Lander\ propulsion}$	300	0	20	20 20	25	25 25	35 125
Orbit (x 1000 n. mi.)	5 x 10	1 x 1.1	1 x 2.3	1 x 3.1	1 x 2.5	1 x 4.1	1 x 7.6
Surface Life Required	8.7 hr.	-	10 min	10 min	2.8 hr	3.5 hr	10 min 5.1 hr

*Released from orbit to achieve γ - 7.5° at 600,000 ft and V_e = 28,000 ft/sec

TABLE 2.6-3. MARS 1969 VOYAGER CONCEPTS

	69-1 Direct Entry	69-2 Direct Entry		69-3 Lander Released From Orbit	69-4 Direct Entry
$W_{injected\ (max.)}$	7030	7030		7030	7030
$W_{arrival}$	6680	6680		6680	6680
$W_{Orbiter}$ (less propulsion)	1420	1420		1420	1420
$W_{propulsion}$ (orbit injection fuel and hardware)	2820	2820		4060	4040
$W_{Lander(s)}$ (less propulsion)	1180 1180	1350 500 500	} Mars '66 Capsules	1130*	1180
$W_{Lander\ propulsion}$	40 40	50 20 20		70	40
Orbit (x 1000 n. mi.)	1 x 5	1 x 5		1 x 7	1 x 1
Launch Window	10 Jan - 9 Feb.	10 Jan - 9 Feb.		3 Jan - 2 Feb.	5 Jan - 4 Feb.

*Ejected at Apofocus to achieve entry at $\gamma_e = 7^o$ and $V_e = 14,500$ ft/sec at 500,000 ft

TABLE 2.6-4. VENUS 1970 VOYAGER CONCEPTS
(ALL DIRECT ENTRY SYSTEM)

	70-1	70-2	70-3	70-4
$W_{injected\ (max.)}$	7260	7260	7260	7260
$W_{arrival}$	6900	6900	6900	6900
$W_{Orbiter}$ (less Propulsion)	900	1250	1250	1250
$W_{Propulsion}$ (orbit insertion Fuel and Hardware)	6000 lbs.	5650	3050	4620
$W_{Lander(s)}$ (less Propulsion)	0	0	1270 1270	500 (Landers or 500 Floaters)
$W_{Lander\ Propulsion}$	0	0	30 30	15 15
Orbit (x 1000 n. mi.)	1 x 1	1 x 2.5	1 x 8.8	1 x 3.8
Lander Surface Life Required	-	-	5.7 hrs.	10 min. or 3.3 hrs.

TABLE 2.6-5. MARS 1971 VOYAGER CONCEPTS

	71-1	71-1A	71-2	71-3	71-4
	Direct Entry	Direct Entry	Direct Entry	Lander Released From Orbit	Lander Released From Orbit
$W_{injected\ (max.)}$	7320	7320	7320	7320	7320
$W_{arrival}$	6950	6950	6950	6950	6950
$W_{Orbiter}$ (less Propulsion)	1420	1420	1420	1420	1420
$W_{Propulsion}$ (orbit injection Fuel and Hardware)	2470	2470	2470	4220	3200
$W_{Lander(s)}$ (less Propulsion)	1480 1480	2960	1950 500 500	1090*	1050* 1050*
$W_{Lander\ Propulsion}$	50 50	100	70 20 20	220	230
Orbit (x 1000 n. mi.)	1 x 1	1 x 1	1 x 1	1 x 1	1 x 7

*Released at Apofocus to achieve entry at $\gamma_e = 7^o$ and V_e = 14,500 ft/sec at 500,000 ft

TABLE 2.6-6. VENUS 1972 VOYAGER CONCEPTS (TYPE I TRAJECTORY) (ALL DIRECT ENTRY SYSTEMS)

	72-1	72-2	72-3	72-4
$W_{injected}$ (max.)	6400	6400	6450	6450
$W_{arrival}$	6080	6080	6130	6130
$W_{Orbiter}$ (less Propulsion)	860	1250	1250	1250
$W_{Propulsion}$ (Orbit Insertion Fuel and Hardware)	5220	4830	2280	3850
$W_{Lander(s)}$ (less Propulsion)	0	0	1270 1270	500 (Landers or 500 Floaters)
$W_{Lander\ Propulsion}$	0	0	30 30	15 15
Orbit (x 1000 n. mi.)	1 x 1	1 x 3	1 x 15	1 x 5
Lander Surface Life Required	-	-	9.5 hrs.	10 min. or 3.9 hrs.
Launch Window	2 Apr. - 2 May	1 Apr. - 1 May	26 March - 25 Apr.	30 March - 29 Apr.

TABLE 2.6-7. VENUS 1972 VOYAGER CONCEPTS (TYPE II TRAJECTORY) (ALL DIRECT ENTRY SYSTEMS)

	72-1 (II)	72-2 (II)	72-3 (II)	72-4 (II)
$W_{injected}$ (max.)	7430	7400	7350	7400
$W_{arrival}$	7060	7030	6980	7030
$W_{Orbiter}$ (less Propulsion)	920	1250	1250	1250
$W_{Propulsion}$ (orbit insertion Fuel and Hardware)	6140	5780	3130	4750
$W_{Lander(s)}$ (less Propulsion)	0	0	1270 1270	500 (Landers or 500 Floaters)
$W_{Lander\ Propulsion}$	0	0	30 30	15 15
Orbit (x 1000 n. mi.)	1 x 1	1 x 2.4	1 x 7.9	1 x 3.7
Lander Surface Life Required	-	-	5.4 hrs.	10 min. or 3.3 hrs.
Launch Window	10 March-9 Apr.	12 March-11 Apr.	18 March-17 Apr.	14 March-13 Apr.

TABLE 2.6-8. MARS 1973 VOYAGER CONCEPTS

	73-1	73-2	73-2A	73-3	73-4
	Direct Entry	Direct Entry	Direct Entry	Lander Released From Orbit	Direct Entry
$W_{injected\ (max.)}$	6100	6100	6100	6100	6000
$W_{arrival}$	5790	5790	5790	5790	5700
$W_{Orbiter}$ (less Propulsion)	1420	1420	1420	1420	800
$W_{Propulsion}$ (orbit injection Fuel and Hardware)	1930	1310	1310	2990	900
$W_{Lander(s)}$ (less Propulsion)	1180 ('69 Landers) 1180	1480 ('71 Landers) 1480	2960 ('71 Lander)	1200*	4000
$W_{Lander\ Propulsion}$	40 40	50 50	100	180	-
Orbit (x 1000 n. mi.)	1 x 2.9	1 x 8	1 x 8	1 x 3	.15 x 9

*Ejected at Apofocus

TABLE 2.6-9. VENUS 1973 VOYAGER CONCEPTS (TYPE I TRAJECTORY)
(ALL DIRECT ENTRY SYSTEMS)

	73-1	73-2	73-3	73-4
$W_{injected\ (max.)}$	6240	6250	6210	6240
$W_{arrival}$	5930	5940	5900	5930
$W_{Orbiter}$ (less Propulsion)	1250	0	1250	1250
$W_{Propulsion}$ (orbit insertion Fuel and Hardware)	4680	390 (Guidance and mid-course Prop)	2050	3650
$W_{Lander(s)}$ (less Propulsion)	0	5550	1270 1270	500 500
$W_{Lander\ Propulsion}$	0	0	30 30	15 15
Orbit (x 1000 n. mi.)	1 x 1	-	1 x 7.1	1 x 2
Lander Surface Life Required	-	?	4.9 hrs.	10 min. or 2.6 hrs.
Launch Window	10 Nov-10 Dec	27 Oct-26 Nov	8 Nov-8 Dec	10 Nov-10 Dec

TABLE 2.6-10. VENUS 1973 VOYAGER CONCEPTS (TYPE II TRAJECTORY) (ALL DIRECT ENTRY SYSTEMS)

	73-1 (II)	73-2 (II)	73-3 (II)	73-4 (II)
$W_{injected}$ (max.)	7440	7360	7420	7420
$W_{arrival}$	7070	6990	7050	7050
$W_{Orbiter}$ (less Propulsion)	1300	0	1250	1250
$W_{Propulsion}$ (orbit insertion Fuel and Hardware)	5770	340 (Guidance and Prop Weight)	3200	4770
$W_{Lander(s)}$ (less Propulsion)	0	6650	1270 1270	500 500
$W_{Landers\ Propulsion}$	0	0	30 30	15 15
Orbit (x 1000 n. mi.)	1 x 1	-	1 x 4	1 x 1.6
Lander Surface Life Required	-	?	3.4 hrs.	10 min. or 2.4 hrs.
Launch Window	30 Oct-29 Nov	25 Oct-24 Nov	29 Oct-28 Nov	30 Oct-29 Nov

TABLE 2.6-11. TRAJECTORY CHARACTERISTICS

	VENUS 1967	VENUS 1968-1969	MARS 1969	VENUS 1970	MARS 1971	VENUS 1972		MARS 1973	VENUS 1973	
						TYPE I TRAJECTORY	TYPE II TRAJECTORY		TYPE I TRAJECTORY	TYPE II TRAJECTORY
Launch Window	12 May-26 June (45 days)	9 Dec-8 Jan	(See Table 2.6-3)	15 Aug-14 Sept	6 May-5 June	(See Table 2.6-6)	(See Table 2.6-7)	19 July-18 Aug	(See Table 2.6-9)	(See Table 2.6-10)
Trip Times, days	175 - 120	154 - 130	270 - 280	121 - 98	210 - 225	122 - 100	190 - 163	205 - 220	~ 118	163 - 143
V_{Entry}, (max.) (at 500,000 ft) ft/sec	35,000	36,600	21,400	37,800	18,900	37,900	37,400	19,200	35,900	36,200
Communications Distance at Encounter (million KM)	90	60 - 70	160 - 190	60 - 70	168 - 180	65 - 77	127 - 140	178 - 245	68 - 102	106 - 116
Arrival Geometry γ_ρ	-50° to -30°	$+20^\circ$ to $+40^\circ$	-16° to -19°	~ - 36	$+9^\circ$ to -14°	32° to 24°	~ -31°	~ -28°	-32° to 0	43° to 20°
ζ_ρ	100° to 80°	40 to 60	95° to 60°	50° to 60°	95° to 73°	40° to 50°	136° to 145°	128° to 105°	52° to 116°	116° to 135°
Sun-Planet-Earth Angle (At Arrival)	108° - 85°	112° - 108°	42° - 37°	73° - 80°	38° - 49°	37° - 60°	~ 76°	39° - 36°	104° - 82°	~ 79°
Planet-Sun-Earth Angle (At Arrival)	32° - 42°	23° - 27°	46° - 51°	23° - 28°	45° - 69°	15° - 26°	50° - 70°	50° - 74°	28° - 47°	47° - 52°
Hyperbolic Excess Velocity (At Arrival) Ft/sec	12,500 - 10,800	12,200 - 14,800	12,200 - 13,880	17,600 - 15,300	~9500	~ 17,900	~16,800	10,000 - 8,200	13,100	14,100

NOTES:

1. γ_ρ - ANGLE BETWEEN THE APPROACH ASYMPTOTE AND THE PLANET'S ORBITAL PLANE

   ζ_ρ - ANGLE BETWEEN THE APPRAOCH ASYMPTOTE AND THE PLANET - SUN VECTOR

2. WHERE EVER TWO VALUES ARE GIVEN FOR A PARAMETER, THE FIRST VALUE OCCURS IN THE EARLY PORTION OF THE LAUNCH WINDOW AND THE SECOND IN THE LATTER PORTION OF THE WINDOW.

3. IN ORDER TO INTERPRET PROPERLY THE SUN-PLANET-EARTH ANGLE AT ARRIVAL, IT IS TO BE NOTED THAT MARS ALWAYS LAGS THE EARTH AND VENUS ALWAYS LEADS THE EARTH

REFERENCES

1. Clarke, V. C. et. al. "Design Parameters for Ballistic Interplanetary Trajectories, Part 1. One-Way Transfers to Mars and Venus," Technical Report No. 32-77, Jet Propulsion Laboratory, January 16, 1963.

2. Bollman, W. E., "Type II Mars Trajectories 1964-1969" Technical Memorandum 312-293, Jet Propulsion Laboratory, March 26, 1963.

3. Unpublished "Mars 1975 Type II Trajectories," Jet Propulsion Laboratory, 1963.

SECTION 3. VOYAGER SYSTEMS

3.1 SUMMARY

To produce a balanced capability in the Voyager system, the following factors were considered: the mission requirements established by the GE Mission Analysis Group, the launch vehicle capability as outlined in the previous section, the balance between Orbiter and Lander scientific equipment weights and mission values in conjunction with the reliability studies, the maximization of delivered scientific equipment with regard to the use of energy-saving eccentric orbits, and the provision for adequate power, guidance and communication capabilities with appropriate application of redundancy at critical junctions in the subsystem functional diagrams.

The final system incorporates one or two Landers, (depending upon the specific mission) and an Orbiter with a single propulsion system for midcourse, terminal, and orbit injection maneuvers. The main communication link from the Lander to the Earth is through a relay in the Orbiter, with secondary communication direct from the Lander to the Earth in the case of the Mars systems. All high data rate communication is transmitted from the Orbiter to Earth through an earth-oriented, ten-foot parabolic antenna. The secondary direct link from Lander to Earth utilizes earth-oriented helix or helical array. Solar cell power supplies are employed in all the Orbiters; primarily because of anticipated restrictions on the availability of sufficient quantities of the desirable radioisotope. However, in the Mars Landers, due to the thermal control requirements during the Martian nights and the long life times required, radioisotope thermoelectric generators are the primary power source. The size of these generators is minimized by using a combination of generator and secondary batteries (the latter for high power demands) and concomitantly, by operating the communication links on an intermittent basis. High volume data storage is provided in thermoplastic recorders both in the Orbiters and in the Landers.

A star tracker, sun sensors, earth sensors, televised photographs of the target planet against the star background, and two way doppler tracking are utilized to provide guidance intelligence. The attitude control system utilizes all gas components for simplicity, light weight, and to take the greatest advantage of currently available hardware.

The single Orbiter propulsion system is a pressurized hypergolic fuel system with combination of radiation and ablative cooling of the thrust chamber.

The basic spacecraft for the program was the Mars 1969 System, which was analyzed in greater detail than any of the others since the prime interest is on Mars exploration and this is the earliest opportunity for which such a spacecraft could be developed.

The logical development of knowledge of the target planets with succeeding opportunities, and concomitantly the expected evolution of sensor and scientific objectives and the variations in the injection and orbit insertion energy requirements, preclude a static Voyager program utilizing invariable spacecraft and payloads. Consequently, an Evolutionary Voyager Program was developed. This program is summarized in

Table 3.1-1, which shows the Mars opportunities from 1969 through 1975 and the Venus opportunities for 1970 and 1972.

It can be seen in this table that the emphasis on the Orbiter is high in the beginning of the Mars program and diminishes to zero in 1975 when a fly-by bus is utilized to deliver the Landers. The increasing knowledge of the biological and geological provinces of Mars obtained by the Orbiters during the 1969 and 1971 opportunities lead to the development of more capability in the Landers for 1973 and 1975 culminating in the detailed survey of possible manned landing sites by a surface rover in 1975.

Due to the cloud cover surrounding Venus, the first Orbiter around the planet would carry radar mapping equipment. Due to the uncertainty in knowledge of the atmosphere and surface conditions on Venus, a minimum Lander of 525 pounds is carried on this spacecraft. In the 1972 Venus opportunity the Orbiter is lighter since the interest in radar measurements of the surface will have waned and a large Lander is carried with considerable capability to make a geological, atmospheric, and possibly biological surveys of the surface environment on Venus.

Two identical Landers were chosen for all the Mars missions since the results given in Section 4 of this volume indicate that when the total weight allocated to the Lander was greater than 1840 pounds, the estimated attainable Mission Value was maximized by dividing the weight between two identical Landers. For the Venus missions, however, one Lander is better because the high weight of the thermal control system causes a severe reduction in the present payload carried by a Lander. This in turn causes the minimum weight for a dual Lander system to be higher than the capability of the Voyager systems.

An analysis was undertaken to determine the influence of a 10 percent reduction in launch vehicle performance on the capability of the Mars 1969 systems. The results indicate that the choice of two Landers is still correct, but each would take a slight reduction in payload or data rate. The most significant change in the Orbiter capability would be the elimination of the color TV pictures. The general conclusion reached was that a 10 percent reduction in launch vehicle performance would not have a serious effect on the capability of the Mars 1969 system.

Orbiters and Landers summarized in the Evolutionary Voyager Program are all based on the weight capabilities of the Saturn C-1B launch vehicle with the S-VI upper stage. Availability and capability of Titan III-C launch vehicle were reviewed and the characteristics of an evolutionary Voyager program utilizing this booster are summarized in Table 3.1-2. Single Landers are indicated in this program because the dual/single Lander crossover point had not been firmly defined at the time the Titan III-C portion of the study was completed. Nevertheless it can be seen in the two tables that the payload capabilities of the Titan III-C systems are substantially equivalent to the Saturn C-1B capabilities. Although the major effort of this study was concentrated on the design of a system for the Saturn C-1B launch vehicle, it is recommended that Titan III-C launch vehicle be seriously considered for the Voyager program. Preliminary comparisons between the Saturn C-1B with the S-VI stage and the Titan III-C launch vehicle show that an all Orbiter Titan III-C vehicle plus Landers delivered to the planet by fly-by buses are more flexible and can accomplish as much or more than the Saturn C-1B system. In addition, preliminary cost comparisons tend to indicate that the Titan III-C systems would be more economical.

TABLE 3.1-1. EVOLUTIONARY VOYAGER PROGRAM

	MARS				VENUS	
	1969	**1971**	**1973**	**1975**	**1970**	**1972**
TOTAL WEIGHT (lbs)	7030	7320	6000	5500	7260	7350
ORBITER (lbs)	2058	2100	1400	1335 (Fly-By Bus)	2145	1800
PAYLOAD	215	223	77	61	137	61
MISSION	S. Hemi. Map (Stereo) Color Characteristics Ionospheric Profile Particles and Field Planetary Emission	N. Hemi. Map (Stereo) Color Characteristics Ionospheric Profile Particles and Field Planetary Emission	Upper Atmosphere Composition Particles and Field	Particles and Fields Planetary Emission	Radar Map, Particles and Field, TV of Clouds, Ionospheric Profile	Particles and Field of Clouds Ionospheric Profile
LANDERS (2) (lbs)	1450 (Each)	2000	2000	2000	525 (Single)	2600 (Single)
PAYLOAD	155 (Each)	255	255	255	60	210
MISSION	Biclogical Analysis, Forms (TV) – Micro and Macro, Geological, Atmospheric	Biological Analysis, Forms (TV) – Micro and Macro, Geological, Ionospheric Profile, Atmospheric	Upper Atmosphere Characteristics, Biological Analysis, Micro and Macro Forms (TV), Geological, Ionospheric Profile, Atmospheric	TV Survey of Possible Manned Landing Areas (Surface Rover) Micro Forms (TV), Geological, Atmospheric	Descent Radar or TV, Surface Hardness, Macro Forms (TV), Atmospheric	Micro and Macro Forms (TV), Geological, Ionospheric Profile, Atmospheric
ORBIT (N. MI.)	1000 x 19,000	1000 x 19,000	200 x 9,000	-	1000 x 4300	1000 x 7300
INCLINATION	55°	45°	53°	-	68°	68°
LIFE						
ORBITER	3 MO.	3 MO.	10 DAYS	-	3 MO.	3 MO.
LANDERS	6 MO.	6 MO.	6 MO.	6 MO.	10–30 MIN.	6.5 HRS.

TABLE 3.1-2. MARS TITAN III-C SYSTEMS

	1969	1971	1973	1975	1977	1979
ALL LANDER SYSTEM						
Weight Injected, lbs.	3000	3000**	2750	2750*	3000*	3000
Lander Weight, lbs.	2230	2230	2000	2000	2230	2230
Trip Time (Days)	275	128	167	~325	(Less Than 1975)	~180
ALL ORBITER SYSTEM						
Weight Injected, lbs.	3350	3600	2800	-	-	-
Scientific Payload in Orbiter, lbs.	223	223	223	-	-	-
Orbit, N. Mi.	1000 x 19,000	1000 x 1000	1000 x 13,000	-	-	-

* Type II Trajectories, but higher than minimum energy trip.

** Higher than minimum energy trip to minimize changes in Lander size.

The transit portions of all missions are essentially the same except for the launch date, trip times, and target planets. After injection into the transit trajectory, the Voyager Spacecraft is separated from the S-VI stage and automatically acquires the cruise attitude, which is confirmed by two way doppler tracking utilizing the high gain antenna. Midcourse maneuver requirements are computed on Earth and transmitted to the spacecraft as is the command for execution. If subsequent tracking data indicates another midcourse maneuver is required, this can be performed; otherwise, the vehicle proceeds on its way to the planet, periodically transmitting the results of any transit science plus engineering and diagnostic telemetry. When the spacecraft is within 2,000,000 n.mi. of the target planet, an image orthicon camera is used to photograph the planet against the star background. This information is used on earth to determine the requirements, if any, for a terminal correction maneuver. This maneuver would be performed approximately 145 hours before encounter (depending upon the magnitude of the error), thus correcting the orbit perifocus altitude to the required degree of accuracy. Additional pictures are transmitted after the manuever in order to determine the accuracy of the resulting trajectory and to provide information for the computation of the Lander separation requirements.

The separation maneuver, planned for 17.8 hours and 150,000 n.mi. before encounter, requires the spacecraft to rotate from the cruise orientation to the direction of the velocity increment to be imparted to the Lander. The Lander is unlatched from the Orbiter and given a small separation impulse by a cold gas system incorporated in the Lander adapter. At a separation distance of 3 feet, cold gas jets are activated, and the Lander is thus spun up and is stabilized prior to the firing of the retro motor. Approximately 17 minutes later when the separation distance has reached 1000 feet, the main solid rocket motor is fired to impart the required velocity increment to the Lander. This separation distance is deemed sufficient to preclude Orbiter degradation or disturbance from the retro system. While the first Lander is moving away, the spacecraft rotates to the attitude for the second Lander velocity impulse. In the case of Mars systems, since the Lander motors are equal and are sized by the higher energy requirement of the out-of-plane case, the second Lander to be separated would be the in-plane Lander. It can be separated as late as 11 hours before encounter and still have enough energy to reach its landing site. This allows time for a repetition of the orientation sequence of the main vehicle if it is disturbed by the separation of the first Lander. During the time between separation and entry, the Landers periodically telemeter engineering data back to the Orbiter using the VHF relay link.

In systems which deliver entry vehicles to another planet, with relatively unknown atmospheres, a prime consideration is the monitoring of the entry phase down to surface impact by extensive diagnostic and scientific instrumentation. If the retardation system fails or surface characteristics are encountered that are outside the design limits, maximum amount of information should be obtained up to the failure point so that future Landers can be modified to eliminate such failures.

In order to maintain communications during this critical period, line of sight between the Orbiter and Lander must be maintained as the Lander enters and descends to the surface. This is accomplished by a judicious choice of the Lander velocity increment. In addition, descent radar incorporated in Mars Landers is used to insure that deployment of the final parachute is delayed until the Lander reaches an altitude of 30,000 feet. This minimizes the descent time, increases the line of sight time, and decreases the required Lander velocity increment. This reduces the descent time for extensive atmospheric measurements, but since the Voyager Landers are to emphasize surface experiments (Mariner B capsules will be designed for atmospheric measurements) this restriction is considered to be reasonable. While the Landers are proceeding toward the planet, the Orbiter has returned to sun orientation, recharged its batteries and as the Landers approach altitudes of 1,000,000 feet, the Orbiter is commanded to assume the orbit insertion attitude. This eliminates an additional attitude orientation maneuver between the end of the line of sight with the Lander and orbit insertion point. After orbit insertion, the orbiter scientific equipment is deployed and communications are begun on the first orbit so that all accumulated Lander information can be transmitted to earth as soon as possible.

3.2 INTRODUCTION

In Section 2.6 of this volume, a number of Voyager concepts based upon the Saturn C-1B capability were identified for the various opportunities. Using these concepts to understand the ramifications inherent in the various types of systems, it is now possible to weigh the alternatives and hence determine the best overall Voyager system.

The major criteria to be employed in this system selection are: maximizing the scientific return consistent with the requirement for higher reliability, the minimum number of modifications between opportunities and the utilization of state-of-the-art components and concepts.

In order to select this optimum system, a number of tradeoffs must be considered:

1. The value of scientific data from an Orbiter versus data from a Lander. As pointed out in Section 1.0 of this volume, the emphasis of the Voyager program is to be on the collection of biological and geophysical-geological data. The question that must be resolved is the relative usefulness of two techniques for obtaining this information.
2. Influence of orbit eccentricity on value of Orbiter scientific data versus increased Lander weight. One of the more important missions that an Orbiter can perform is obtaining TV or radar maps of the surface. The best orbit for obtaining this information would obviously be a low circular one; however, since circular orbits are quite costly from the standpoint of propulsion systems weight required, larger Landers could be obtained with elliptical orbits if it were judged to be worth the decrease in the quality of the map.
3. Advantages of carrying the Lander into orbit versus weight penalty involved. As the Voyager concepts given in Section 2.6 indicate, it is extremely costly in propulsion system weight to carry the Lander into orbit before it is released. Therefore, in order to choose this mode of operation, the scientific payoff must be large.
4. Greater variety of scientific information versus a higher reliability for a lesser variety. As the preliminary concepts indicate, rather large Landers (2000-3000 lbs.) may be possible, especially with elliptical orbits. Since this represents a significant amount of payload, the possibility of utilizing two identical Landers to increase the reliability of obtaining the more valuable data must be explored.
5. Order in which scientific data should be obtained versus energy changes with opportunities versus common system size. If one were to consider only the first opportunity in designing the Voyager system, chances are high that the system would be rather poorly designed for obtaining other information during the later opportunities. Therefore, consideration must be given to the order in which it would be desirable to obtain the scientific information in order to design the system so as to minimize the number of changes (except for payload) that would be required for succeeding opportunities.
6. Type I versus Type II trajectories. As the Voyager concepts given in Section 2.6 indicate, the choice of Type I or Type II trajectory was not immediately evident. In a number of cases, the Type II trajectories

yield greater payload capability but at the expense of longer trip times. In order to choose the proper trajectory, the trade-off between this greater payload capability and trip time in terms of system reliability is considered in Section 4. However, another factor which must be taken into account is whether this increased payload is required to fulfill the evolutionary scientific program. The results that are obtained from considering these various trade-offs are a definition of the Voyager system in terms of mission profiles and a Voyager Evolutionary Program.

As discussed in Volume VI, a review of the Venus 1967 opportunity indicates that an attractive Voyager system cannot be developed in time to meet the May 1967 launch date. Therefore, the first opportunity for the Voyager system is Mars 1969. The Venus 1968-69 opportunity was not considered since development of a system for this opportunity would interfere with the development of the more important Mars system. Other opportunities through 1975 were considered for both Mars and Venus.

One of the factors that should be considered in arriving at the most attractive Voyager system is its growth potential. Therefore, a brief analysis was conducted to determine the general characteristics of a 60,000 pound spacecraft and to hypothesize possible modes of operation.

3.3 MARS 1969 SYSTEM

As indicated in Section 1.0, it would be very desirable to obtain a broad coverage of the terrain and color features of the Mars surfaces on the early missions to identify interesting areas for more detailed surface exploration. Therefore, it was decided to employ a TV mapping Orbiter during this opportunity.

Since a number of interesting areas on the surface have been identified (Section 1.5) for Earth observations and since early Lander design studies indicated that no strong design advantages would be gained by carrying the Lander into orbit, the direct entry approach was chosen. Therefore, the question that must be resolved is: what is the best eccentricity for the orbit from an overall system consideration? Since, as indicated in Table 2.6-3, the Lander weight could be more than doubled with the use of eccentric orbits, the incentive is high to utilize such an approach.

3.3.1 ORBIT ECCENTRICITY

With the nominal orbit periapsis altitude established at 1000 nm to preclude contamination of Mars by the unsterilized Orbiter, and the inclination established by geophysical experiment criteria, the remaining unknown in the orbit decision was the eccentricity or apoapsis altitude. This problem was approached by reviewing the orbiter mission requirements. Since the objective of acquiring a television map of Mars surface was a dominant consideration in the data transmission (and therefore power supply requirements), a study of the effect of the TV mission on the orbit selection was undertaken.

A. Effect of Eccentricity on TV Resolution

It was recognized that the scientific community desires the finest photographic resolution possible. However, preliminary estimates of expected data rates to be realized by a practical communication system indicated that any complete survey of the surface that could be transmitted in a reasonable orbiter life through the communication distance of approximately 1.29AU at encounter would be restricted to resolution of approximately 500-2000 meters. (All resolutions will be discussed in terms of optical resolution of 2 line pairs with dark or light discernible between the lines.) Therefore, it was decided that the complete map would be executed at one resolution and that additional optical and television camera equipment would be carried to realize higher resolutions over certain portions of the surface of Mars. Since eccentric orbits were being considered, there was the inherent penalty that a resolution obtained at perifocus would be degraded on each side of orbit perifocus as the altitude above the surface of the planet increased.

B. Idealized Orbit Geometry

In order to obtain TV photographs, the orbit insertion, at periapsis, should occur in daylight. The orbit periapsis can occur over the noon meridian for the approach geometry for the Mars 1969 mission. With Mars at winter Solstice and the line of nodes normal to the planet-sun line, the Orbiter will cross each terminator and the equator at the same altitude. This symmetrical orbit geometry was used to simplify the TV mapping analysis, even though this is precisely true for only one day during a mission because of the effects of orbit precession and seasonal progression, and may not occur on some missions because of variable arrival geometry through the launch window. At any other day, periapsis will not coincide with the noon meridian and the Orbiter will cross the terminators at unequal altitudes.

C. Orbit Inclination

Because Mars is a rotating planet, it is desirable to have a near polar orbit in order to map as large a portion of the planet surface as possible. However, placing an artificial satellite equipped with a communication system in orbit around Mars presents a unique opportunity to obtain planetological information on Mars by using doppler techniques to determine the perturbations of the orbit as time progresses. The effect of Mars equatorial bulge could be detected if the maximum inclination of the orbit plane to the Mars equatorial plane was set at 60°. To afford some margin for guidance error, the orbit inclination was set at 55° with injection in the southern hemisphere, which is the more interesting of the two hemispheres.

D. Map Area

Of course, in an inclined orbit, one cannot map an entire hemisphere because the Orbiter will not pass over the pole of the rotating planet. The 55° inclination chosen for our study allows a map to be made of ~80% of one hemisphere of Mars or ~40% of the entire planet.

E. Swath Area

As the orbiter proceeds along an eccentric orbit with the symmetrical geometry described in Section 3.3.1-B above, a television camera looking along the local vertical would photograph a swath on the ground along the orbit track. The varying altitude of the eccentric orbit would cause the width of the swath to vary in proportion to the altitude because the constant angle of the camera's field of view would be intercepted at varying distances from the camera.

The field of view of the camera is a function of the ratio of orbiter altitude, to perifocus altitude, perifocus resolution, and the number of lines available in the vidicon. The 500 lines of the vidicon were reduced by 30% for the Kell factor (to 350) and by 10% for overlap (to 315). Optical resolution is half of the television resolution. Periapsis optical resolutions from 624 meters to 2000 meters were included in this study. Due to the narrow view angle of the camera optical system, a flat planet was assumed at the intersection of the view angle of the camera and the surface of the planet. Swath width is calculated on the basis of a vidicon television camera aligned with the local vertical. Swath length is the length of the great circle from terminator to terminator on one half the circumference of the planet. The effect of the rotation of Mars was not included because the orbit requires approximately 110 minutes to proceed from terminator to terminator and since the planet Mars turns only 26.8 degrees during this time period, the resulting increase in swath length is relatively insignificant. Orbits from 1000 and 5000 nm to 1000 and 29,000 nm were considered in this analysis. Area obtained from each orbit was calculated on the basis of a trapesoidal approximation of the double concave lense shape of each swath.

F. Map Completion Time

The number of orbits required to complete a map of the available area of the hemisphere (due to orbit inclination) is approximated by dividing that area by the swath area.

This is an approximation because orbit inclination causes swath crossing resulting in wasted frames. As the map becomes almost complete, swaths become less efficient because of the excessive overlap when a swath is laid over the flower petal shaped area remaining to be photographed between previously acquired swaths.

The time to complete a map is the number of orbits times the orbital period.

Figure 3.3.1-1 shows the time required to complete the hemisphere map in earth days for various orbit eccentricities with respect to perifocus resolution in meters. Cross plots of this curve produce Figure 3.3.1-2, time required to completely acquire a map for various resolutions with respect to apoapsis altitude. In Figure 3.3.1-3, periapsis resolution and number of frames per orbit, it can be seen that effect of orbit eccentricity is very minor because of the small difference in altitude when crossing a terminator.

In Figure 3.3.1-4, total frames per map are plotted for various orbit eccentricities with relation to the resolution at perigee. It is seen that the change in eccentricity is not a strong factor in the total frames per map. However, when plotting data accumulation rate and frames per day versus resolution at perifocus in meters, we see that the large change in orbit period over the range of orbit eccentricities studies is a strong factor in the rate of data accumulation. Orbits of higher eccentricity reduce the data rate requirements.

Figure 3.3.1-5, data rate, is plotted on frames per day versus apoapsis altitude and 1000 nm for various resolutions at periapsis. It can be seen on this curve how the nominal orbit finally chosen, 1 x 19, is near the knee of the curve as far as data rate is concerned.

It can be seen that for any orbit eccentricity, there is an optical resolution that will permit the map to be completed in a reasonable length of time, i.e., 20-40 days. In the absence of a firm mapping resolution requirement, TV map resolution was not the dominating factor in setting orbit eccentricity.

G. Lander Weight

Lander weight is plotted with respect to orbit eccentricity for various net orbiter weights in Figure 2.5-17 in Volume II. The gain in Lander weight is quite small for apofocus altitudes higher than 19,000 n.m. This curve was instrumental in setting the apoapsis altitude at 19,000 n.m.

H. Orbiter-Lander Line of Sight

Another factor, which is more fully defined later in the study, is the operation of the Lander-to-Orbiter relay link. An extremely long orbit period would reduce the capability of such a relay link by reducing the frequency of occurrence of the line-of-sight conditions or opportunities that are required for a communication link.

Since the data rate for a communication link is proportional to the square of the distance, the most effective line-of-sight opportunities for an eccentric occur when the Orbiter is near periapsis. A particularly ineffective relay link situation occurs when the orbit period is close to 3/2 times the Mars rotational period of 24.6 hours. Line-of-sight opportunities would then occur at intervals of 74 hours.

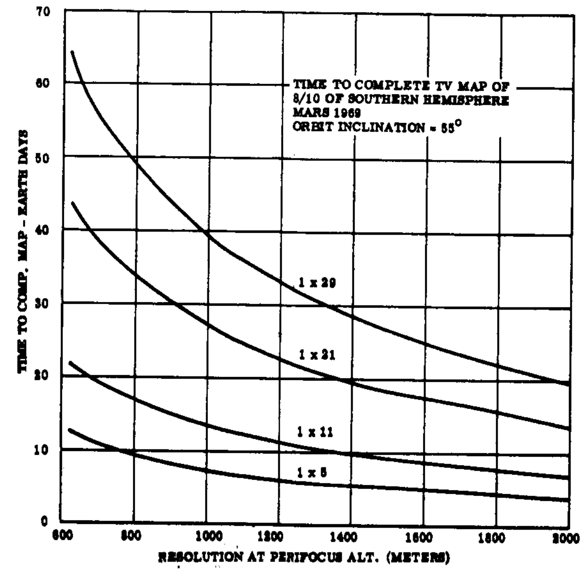


Figure 3.3.1-1. Time to Complete TV Map of 8/10 of Southern Hemisphere

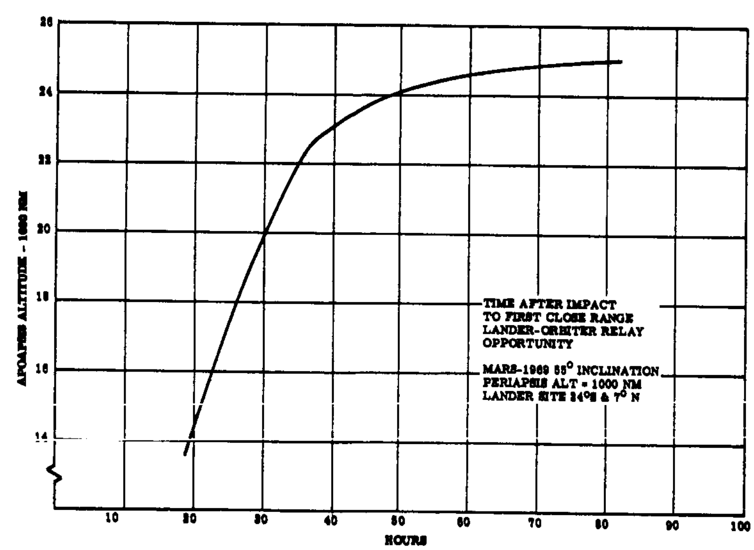


Figure 3.3.1-2. Time After Impact to First Close Range Lander-Orbiter Relay Opportunity

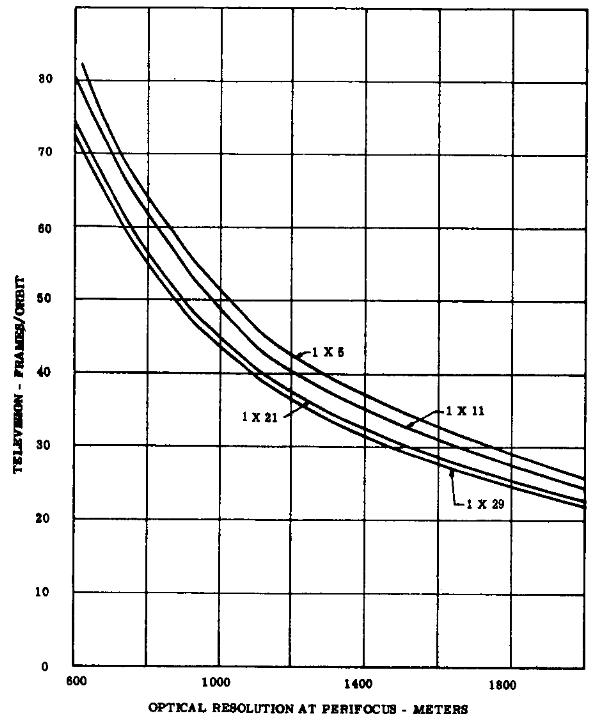


Figure 3.3.1-3. Periapsis Resolution and Number of Frames per Orbit

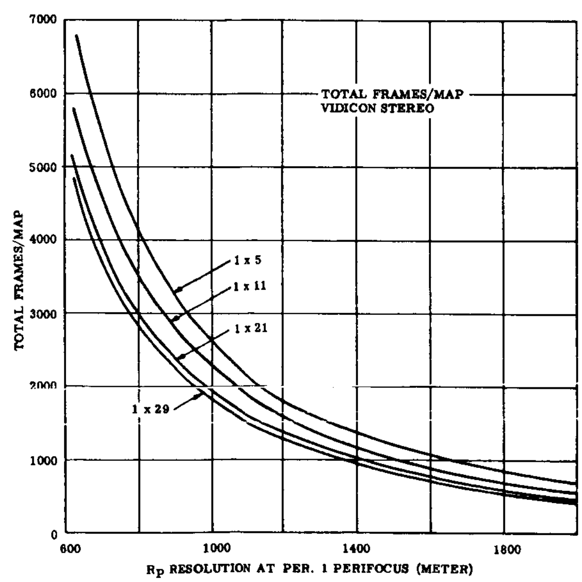


Figure 3.3.1-4. Total Frames/Map, Vidicon Stereo

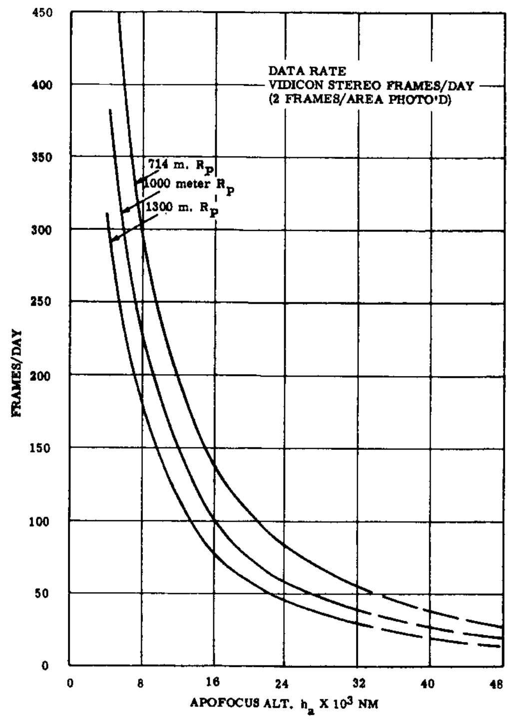


Figure 3.3.1-5. Data Rate, Vidicon Stereo Frames/Day

Expected guidance error will cause variations in the orbit apoapsis altitude and eccentricity. The maximum possible apoapsis altitude for a nominal 1 x 19 orbit is predicted to be 25,000 nm with a 3 σ probability. This orbit has a period of 37 hours which is nearly equal to 3/2 x 24.6 or 36.9 hours. Since this is the 3 σ outer limit of probability, the penalty on relay link operation is reluctantly accepted.

Later in the study, plots of line-of-sight times and ranges become available for various orbits and the two selected Lander sites. It can be seen that relatively long range relay windows will occur near the fifth hour of Lander operation for any orbit for the 7° N. latitude site, thus yielding some data returns very early in the Lander operational life, which is desirable from the reliability or get-it-before-it-fails point of view. First returns are available from the 24°S latitude Lander site slightly later than five hours after impact and at slightly longer ranges.

Subsequent to the first relay data returns, the close range line-of-sight opportunities that are ~ 25 times more effective for high data rates will be utilized. A plot of hours of an operation (impact = zero) to the first close range relay with respect to orbit apoapsis altitude is shown in Figure 3.3.1-2. The time to first close range relay period is very high for a 1 x 25 orbit, i.e., 80 hours, and the curve is very steep in the direction of rapidly increasing times. The 1 x 25 orbit should, indeed, be the maximum expected eccentricity even for the 3 σ edge of the distribution curve.

I. Selection of Eccentricity

In view of the above criteria, it was concluded that a nominal 1 x 19 (1000 nm) orbit was the best compromise for the Mars 1969 opportunity affording almost the maximum practical total Lander weight with reasonable map resolution and map completion times and acceptable lander-to-orbiter relay link opportunities.

3.3.2 TELEVISION SUBSYSTEM

A. Resolution and TV Camera Selection

Referring to Figure 3.3.2-1 and 3.3.2-2, it can be seen that a periapsis resolution of 1000 meters is indicated for a hemisphere map completion time of 25 days for the selected nominal 1 x 19 orbit (1000 n.mi.). This is the optical resolution that is planned for the complete map of the available portion of the hemisphere. Orbit variations due to guidance error are expected to range from a minimum of 1 x 14.5 (1000 n.mi.) to a maximum of 1 x 25 (1000 n.mi.) where perifocus altitude is 1000 n.mi. and apofocus altitude ranges from 14,500 n.mi. to 25,000 n.mi. The map completion times for 1000 meter optical resolution at perifocus for these extreme orbits are 16 and 35 earth days, respectively. Vidicons are used for the mapping cameras. The higher sensitivity of image orthicon cameras was employed for both color and high resolution photographs. The physical size of the optics of the high resolution camera, a strong determinant of the size of the PHP, and the weight of this optical system, set a practical limit of 20 meters on the high resolution camera. Intermediate resolution, based on equal ratios from 1000 meters to 20 meters, was set at 140 meters. Therefore, three image orthicons with the appropriate filters (red, yellow-green and blue provide color pictures by superimposing the three medium (140 meters) resolution pictures. In order to provide the maximum probability of locating an area photographed in successive resolutions, it was decided that "nested" sets of one high resolution picture and one medium resolution color picture would be provided. Four (4) image orthicon cameras are exposed at the same instant to provide these nested sets.

B. Stereo Mapping

Vertical resolution of 345 meters of landscape features is provided by the stereo mapping subsystem. Stereo mapping is accomplished in the Orbiter by using two vidicon cameras installed in the PHP at fixed angles, 19 1/2°, to the vertical axis of the Planet Horizontal Package. The PHP is always pointing to the center of the planet when the orbiter is in orbit. A layout was made, Figure 3.3.2-3 in order to study the operation of the stereo mapping subsystem. The layout is based on the nominal orbit of 100 x 19,000 n.mi. with 10% overlap on the leading stereo camera. It can be seen from Figure 3.3.2-3 that one of the effects of variable altitude is the gradual decrease in the angle between the local horizontal and the line of sight from the camera. When this zenith angle becomes zero, of course, the camera is looking at the horizon of the planet from the orbiter and surface definition is lost. Consequently, the stereo cameras are not effective on the outer portion of the mapping pass. A nadir vidicon, therefore, is employed which is always looking straight down from the Orbiter. This camera is used to fill in the gaps that cannot be covered by the stereo cameras and operates from station No. 17 through 26 providing 10 frames per half of mapping pass. At station number 22 the leading stereo cameras' picture area had arrived at the 90° point or terminator for the idealized orbit, and the angle between the local horizontal and the line of sight of the cameras is approximately 40°. Therefore 22 stations are utilized on the leading stereo camera. The lagging or trailing stereo camera begins to have a much higher degree of overlap when it is used at the same stations as laid out for the leading stereo camera which is made on the basis of 10% overlap. Gradually increasing spacing can thus be used on the stations 14 through 22, and 21 lagging stereo frames are provided per half of the pass. There will be two stereo frames at the zero or perifocus station.

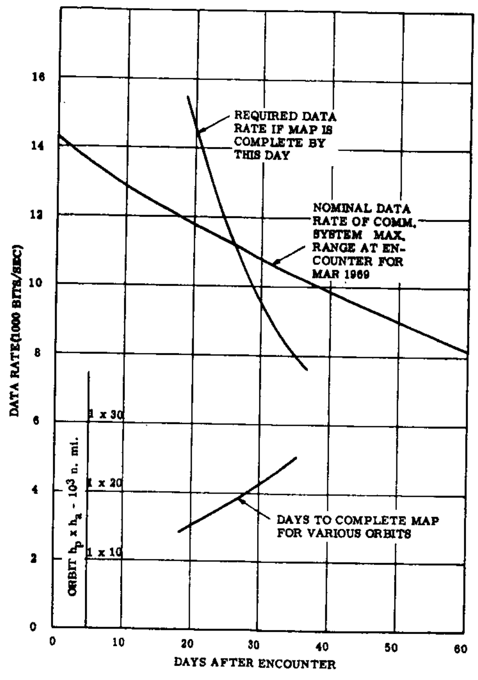


Figure 3.3.2-1. Data Rate Versus Days After Encounter

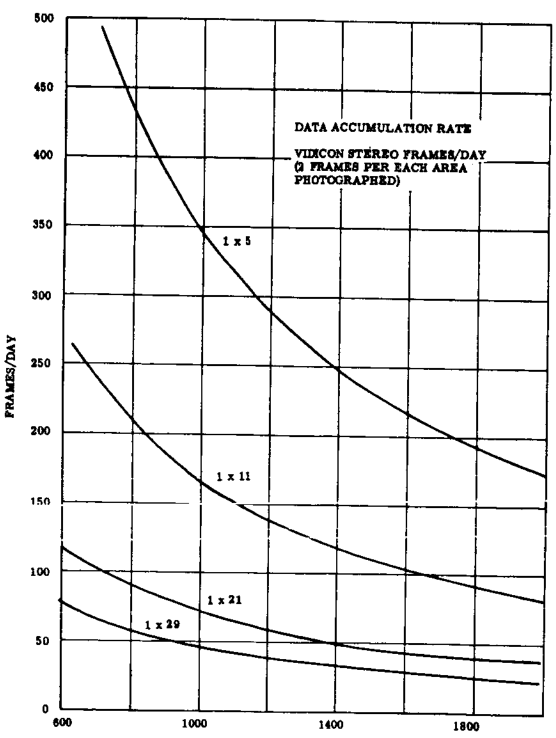
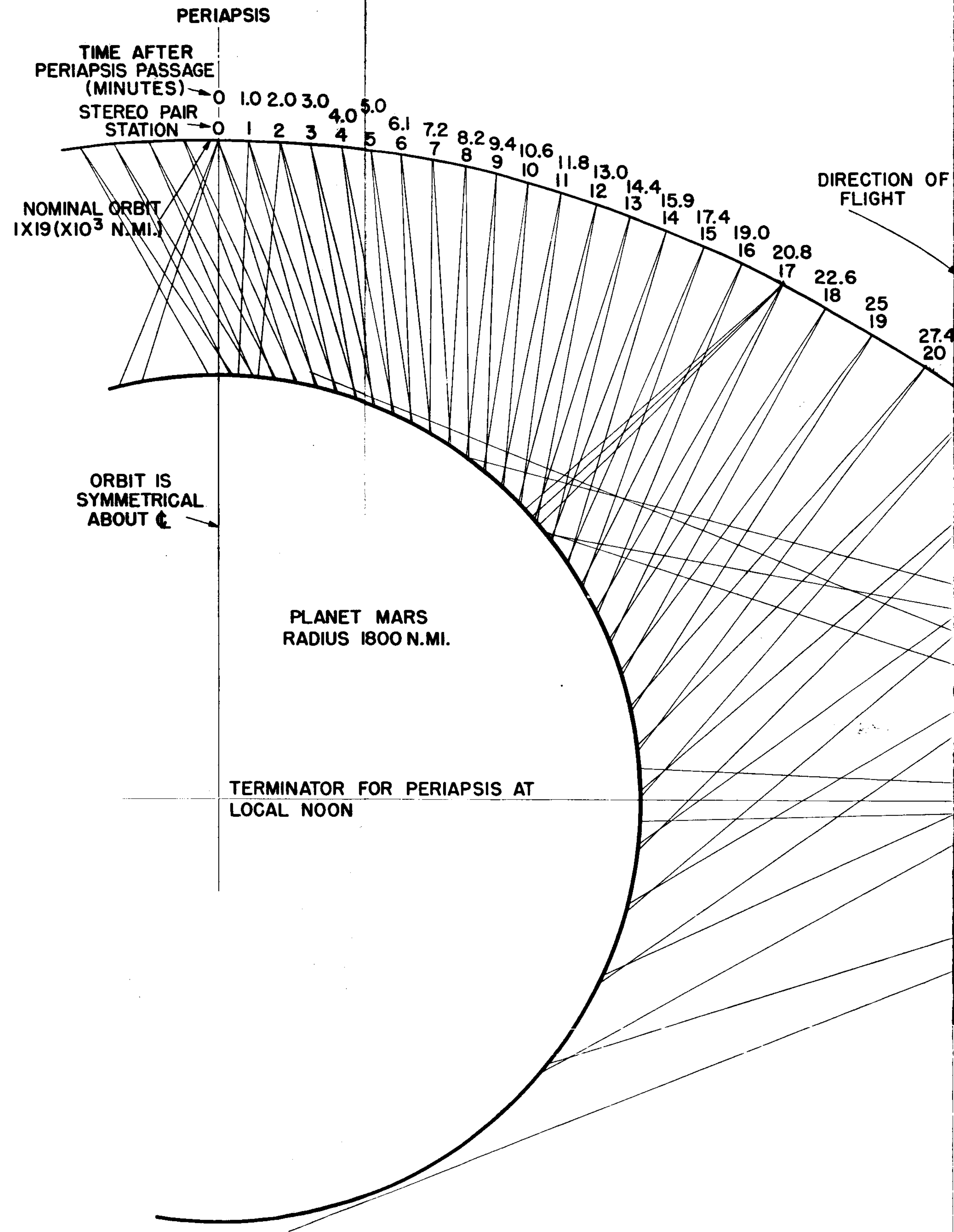


Figure 3.3.2-2. Data Accumulation Rate



PERIAPSIS
TIME AFTER PERIAPSIS PASSAGE (MINUTES)
0 1.0 2.0 3.0 4.0 5.0 6.1 7.2 8.2 9.4 10.6 11.8 13.0 14.4 15.9 17.4 19.0 20.8 22.6 25 27.4
STEREO PAIR STATION
0 1 2 3 4 5 6 7 8 9 10 11 12 13 14 15 16 17 18 19 20
DIRECTION OF FLIGHT
NOMINAL ORBIT 1X19 (X10³ N.MI.)
ORBIT IS SYMMETRICAL ABOUT ℄
PLANET MARS RADIUS 1800 N.MI.
TERMINATOR FOR PERIAPSIS AT LOCAL NOON

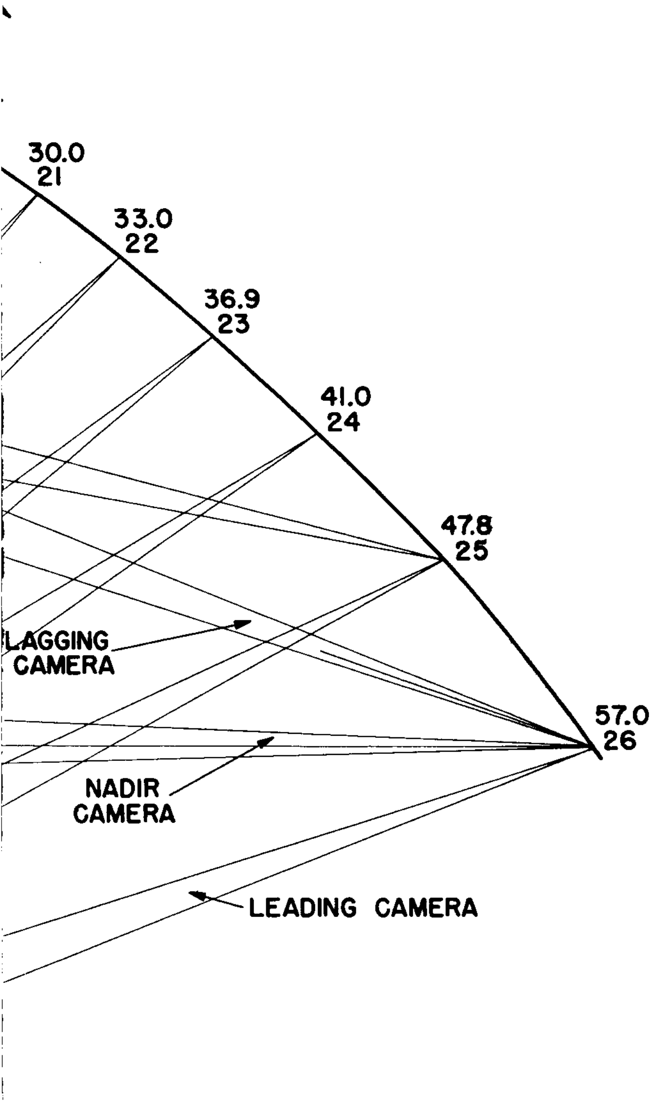


Figure 3.3.2-3. Mars 1967 Orbiter (Layout of Stereo Mapping Mission)

The total number of vidicon (stereo) frames per orbit is therefore 108 which is approximately equivalent to 52 stereo pairs per orbit due to the increased overlap encountered by the lagging stereo camera. Four image orthicon frames are taken for each of 52 stereo pairs. The 208 1.0 frames and the 108 vidicon frames establish the nominal rate of 316 frames per orbit. The 108 vidicon frames at 1.1×10^6 bits per frame provide $.119 \times 10^9$ bits per orbit.

C. Data Rates

Since it is not convenient to transmit television information direct to earth during that portion of the orbit in which the pictures are being acquired, the total amount of information stored per orbit is a 1.03×10^9 bits. This determines data storage volume requirements. It can also be seen that the time interval between stations varies throughout the mapping portion of this orbit from a minimum of 60 seconds at perifocus to a maximum of 9.7 minutes at the 90^o central angle. The requirement of exposing the 4 image orthicon cameras at the same time to provide concentric medium and high resolution pictures sets the recording rate requirements.

D. Data Recorders

The data storage equipment records 2 vidicon frames @ 1.1×10^6 bits per frame and 4 I.O. frames @ 4.4×10^6 bits per frame in the minimum time interval of 1 min. between the perifocus and the no. 1 picture station. This amounts to 19.8×10^6 bits per minute or 330 kilobits per second. The maximum practical recording rate of a thermoplastic recorder is about 200-250 kilobits per second. Therefore, two (2) TPR's are provided in the Orbiter in order to record the television data acquired in the minimum time interval between stations.

The storage volume of each TPR is set at 1×10^9 bits because there are no significant weight savings for a smaller volume in a TPR. The major portion of the weight of a TPR is invested in the electronics, and indexing mechanism. A reduction in volume only reduces the number of plastic coated glass plates. Even if one of the two TPR's fails to function the remaining one would have sufficient storage volume to record all the television information that could be acquired on one orbit, i.e. 1×10^9 bits.

The loss of recording rate capability due to the failure of one of the TPR's will cause the ratio of color and high resolution picture sets to stereo pairs to be reduced. In the perifocus region, the two stereo vidicons would be exposed on schedule and would be recorded in 11 seconds and the four (4) I.O.'s (color and high resolution) would require 88 seconds to be recorded on one (1) TPR at 200 kilobits per second. The total recording time of 99 seconds is thus greater than the minimum station time interval of 60 seconds. Therefore, the ratio of color and high resolution picture sets to stereo power must be reduced to 1 to 2 in the event of failure of one (1) TPR. This ratio would be held constant until station no. 18 when the interval has increased to 2.2 minutes.

E. Effect of Synchronism on Mapping Mission

Errors in vehicle attitude during Orbiter insertion propulsion burn, magnitude of the velocity increment, and periapsis altitude will cause a variation in apoapsis altitude. The nominal orbit has a periapsis altitude of 1,000 n.m. and an apoapsis altitude of 19,000 n.m. The maximum expected apoapsis altitude is 25,000 n.m. and the minimum expected apoapsis altitude is 14,500 n.m. The maximum expected orbital period is 37 hours, the nominal for the 1 x 19 orbit is 27.3 hours and the minimum expected period for the one x 14.5 orbit is 20 hours.

Thus it can be seen that the expected range of orbital periods includes the 24.6 hours rotational period for the planet Mars. In this case the orbit track would repeat itself every orbit and the television equipment could only acquire one swath. Other such periods occur with integral ratios of the planet rotational period and the orbital period. These synchronous spikes are shown in Figure 3.3.2-4.

Table No. 3.3.2-1 shows the number of situations that can occur. Since it takes about 23 orbits to complete the nominal map, synchronous ratios involving more than 24 orbits will not degrade the mapping mission. If such a situation did occur, it would mean that the orbiter would be in position to repeat its track at the end of the mapping mission, and therefore it would be much easier to obtain additional medium and high resolution photographs of areas identified on previous passes. The attached graph shows the effect of some of these spikes in terms of percentage of coverage at the equator in 30 days versus the orbital period in hours. It is seen on the curve that the probability of acquiring an orbital period that causes a serious deterioration of the mapping mission is quite low.

Since these periods occur within the expected guidance error in our orbit insertion maneuver, it is necessary then to provide some method of altering the period of the orbit in order to move off one of the spikes. The equatorial surface speed of Mars is 460 n.m. per hour, and the swath width along the equator is nominally 274 n.m. If a synchronous orbit is achieved, the maximum change in the orbital in hours period that must be produced by a trim rocket or other method of changing the orbiter velocity is the ratio of swath width in n. miles to the equatorial speed in n.m. per hour. This amounts to 6/10 of an hour in this worst case. This can be achieved by an impulsive firing at periapsis of approximately 16 feet per second. The method proposed is to use expulsion of the helium gas used to pressurize the fuel tanks. This orbit-trim operation would occur after the orbit has been carefully determined by tracking for a period of a week or two after encounter.

If the orbit that is realized coincides with the synchronous "spike" and if the orbit trim operation fails to function, the effect on the mapping mission can be diminished by employing the alternate mode of controlling the pointing attitude of the Planet Horizontal Package. After the orbit is determined precisely, by two-way doppler tracking of the orbiter for several orbits, the standard television mapping sequence can be executed for the number of orbits that will present new areas underneath the orbiter to be mapped. At the end of this period, the mode of pointing the PHP will be changed from the horizon sensing equipment to a commanded mode. The two required hinge angles between the orbiter body and the PHP are then calculated on earth for each television mapping station along the orbit. The PHP is then commanded to look to one side or the other of the path as the orbiter flies over the surface of Mars that has already been photographed. There is some photographic degradation due to the additional side angle; however, this mode greatly minimizes the effect of a synchronous orbit situation.

If it is assumed that good mapping pictures can be obtained by deflecting the PHP so that one additional swath is obtained on each side of a previously acquired orbit track, then the number of orbits to complete the map can be reduced to eight (8), (24 orbits for the original mapping requirement divided by 3), the number of critical ratios is now greatly reduced and these ratios lie to the left of the dashed line in Table No. 3.3.2-1

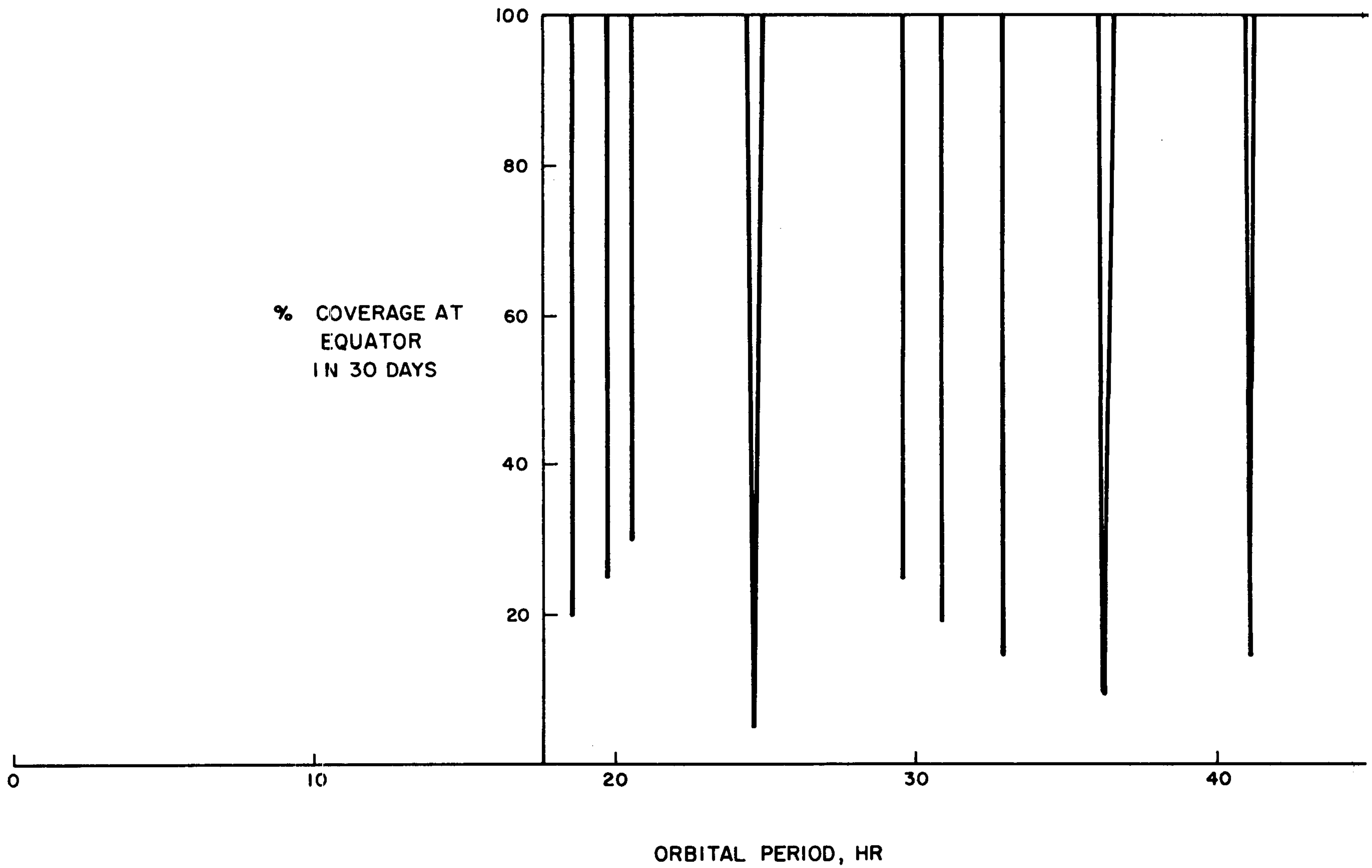


Figure 3.3.2-4. Mapping Coverage (for Nine Worst-Case Orbit Periods Only)

TABLE 3.3.2-1. POSSIBLE SYNCHRONOUS ORBITAL PERIODS (ORBITS/DAY) RESULTING IN LESS THAN 100 PERCENT COVERAGE WHEN USING NADIR TRACKING PHP

M	N/M											
1	1											
2	1/2	3/2										
3	2/3	4/3										
4	3/4	5/4										
5	3/5	4/5	6/5	7/5								
6	5/6	7/6										
7	4/7	5/7	6/7	8/7	9/7	10/7						
8	5/8	7/8	8/8	11/8								
9	5/9	7/9	8/9	10/9	11/9	13/9	14/9					
10	7/10	9/10	11/10	13/10								
11	6/11	7/11	8/11	9/11	10/11	12/11	13/11	14/11	15/11	16/11		
12	7/12	11/12	13/12	17/12								
13	7/13	8/13	9/13	10/13	11/13	12/13	14/13	15/13	16/13	17/13	18/13	19/13
14	9/14	11/14	13/14	15/14	17/14	19/14						
15	8/15	11/15	13/15	14/15	16/15	17/15	19/15					
16	9/16	11/16	13/16	15/16	17/16	19/16	21/16					
17	9/17	10/17	11/17	12/17	13/17	14/17	15/17	16/17	18/17	19/17	20/17	21/17
18	11/18	13/18	17/18	19/18								
19	10/19	11/19	12/19	13/19	14/19	15/19	16/19	17/19	18/19	20/19	21/19	
20	11/20	13/20	17/20	19/20	21/20							
21	11/21	13/21	14/21	16/21	17/21	19/21	20/21					
22	13/22	15/22	17/22	19/22	21/22							
23	13/23	14/23	15/23	16/23	17/23	18/23	19/23	20/23	21/23			

M = integral number of days to map completely with no overlap at equator

N = integral number of orbits completed in M days

$N = \frac{N}{M}$ = average number of orbits/day $\frac{T_p}{T}$

$1 \leq N \leq \frac{21000}{500} = 42$

$1 \leq M \leq \frac{42}{.55} \approx 77$

$.55 \leq N \leq 1.4$

$.5 \leq N \leq 1.5$

Possible Synchronous Orbital Periods (Orbits/Day) Resulting in Less than 100 Percent Coverage When Using Nadir Tracking PHP

Orbital Periods for which 100 percent coverage is not possible by tilting PHP

Orbital Periods for which tilting PHP results in 100 percent coverage

F. Initial Operation of TV Mapping Subsystem

Due to the expected variation in orbital apafocus altitude and period, the television subystem cannot be commanded to commence operation at a certain time interval after orbit insertion. The time over the illuminated surface of Mars is approximately 110 minutes. This length of time is much shorter than the expected variation in orbital period (i.e., from 37 hours to 20 hours).

Even using the nominal period of 27.3 hours as the basis for timing the television sequence could cause the sequence to be attempted completely outside of the limits of the orbit segment over the illuminated surface.

A photometer must be provided on the PHP that can sense when the orbiter crosses the terminator. This signal can be used to initiate the television mapping sequence.

In the idealized stereo mapping mission layout, Figure No. 3.3.2-3, the 90° central angle point on the orbit coincides with the terminator and the equator. In this particular mission, the terminator signal would initiate the photographic sequence starting with the nadir vidicon at Station 26 and proceeding through the sequence at the varying, prescheduled, time intervals between stations.

But, this idealized orbit geometry is possible for only one particular day in the launch window. The intersection of the orbit plane and the terminator may not coincide with the line of modes and the equator. Periapsis may lie far off the noon meridian. The orbit inclination may be far different than the planned inclination. Therefore, the terminator would not coincide with the 90° central angles of the orbit and the television mapping sequence would not start with Station No. 26.

Although the expected variation in orbital periods is large, (a difference of 17 hours from the lowest to the highest), the difference in duration of the mapping segment of the orbit is much smaller. For example, the time for the orbiter to move from periapsis to the 90° central angle point on the 1 + 14.5 orbit is 54 minutes and on the 1 + 25 orbit is 55+ minutes.

This small variation indicates that the orbit period need not be known in order to program the television mapping sequence. Only the perifocus location and orbit inclination is required and these elements are determined by the approach geometry of the spacecraft trajectory. The approach geometry can be accurately determined from the planet based on information obtained in the terminal guidance observation and can be rechecked after execution of the terminal correction maneuver. This information will enable the earthlings in the Voyager control center to choose the appropriate initial television sequence station and to transmit these commands to the orbiter where it can be placed in the memory of the command subsystem. On the first orbits, the terminator signal received by the light sensor on board the orbiter will cause the television subsystem to begin operations at the preselected station. This station could be either earlier than No. 26, such as 22, 23, 24, or later stations such as 27, 28, 29. Swaths photographed in early portions of the orbiting mission will then be properly spaced with relation to the orbiter altitude, and will provide the required degree of overlap.

3.3.3 ORBITER COMMUNICATION SUBSYSTEM

A. Required Data Rate

Since during the Orbiter mission, Mars is gradually receding from earth, the nominal communication data rate is decreasing in proportion to the square of the distance between the two planets. The constant data rate accumulation for the selected orbit and mapping resolution is 1.03 x 10^9 bits per orbit. The high electrical power requirements of the orbiter-to-earth high rate data telemetry link, even though transmission is through the high gain parabolic antenna precludes operation of this data link when the orbiter is acquiring stereo mapping, color, and high resolution pictorial information during the mapping portion of the orbit. This loss of communication time is a small part, 18 hours, of the total orbital period of 27.7 hours. The accumulated data from one pass, 1.03 x 10^9 bits must be transmitted in 27.7-1.8 or 25.9 hours. The required nominal data rate is 11.1 kilobits per second. The telemetry link must have this capability at the completion of the mapping period or on the 26th day after encounter for the 1 x 19 (1000 n. mi.) orbit and 1000 meter optical perifocus resolution. The Mars/Earth distance, 1.29 AU, is the maximum at encounter for the spacecraft launched on the last day of the launch window. The ratio between the 26-day rate and encounter communication rate is 1.285; therefore, the required data/rate at encounter is 14.25 kilobits per second.

B. Antenna Size and Power Level

It had been determined by the communications section that 70 watts radiated through a 10-foot diameter parabolic antenna would produce a rate at this encounter for the Mars 1969 mission of 22.2 kilobits per second. Since the data rate is proportional to the square of the diameter of the parabolic antenna and is directly proportional to the radiated power, then the product of the antenna diameter squared and the power level would be proportional to the data rate. For a rate of 22.2 kilobits, 7,000 ft^2 watts are required. The 14.25 kilobit rate requires 4,490 ft^2 watts for a ten foot diameter dish. When optimizing the power level and dish size, the orbiter constant 0 pointing error of $\pm 1^o$ was utilized in calculating pointing loss and required power for each dish size in the optimization study. The efficiency for various sizes of the klystron power amplifier was varied in accordance with an estimated curve. The power regulation efficiency was assumed to be a constant 75%. The weight of the power supply, which in this case was solar cells, was 2.66 watts per pound. This is the specific weight of solar array for the extended fixed solar array at the outer perimeter of the orbiter. Since we were varying only communications system power level in the orbiter the lighter in-board solar area panels were allocated to the other orbiter power requirements. Weight of the antenna and actuator was estimated at .39 times the square of the diameter in feet. System weight is the weight of the parabolic antenna and the power supply. Results are tabulated in Table 3.3.3-1.

TABLE 3.3.3-1. POWER SUPPLY AND PARABOLIC ANTENNA OPTIMIZATION FOR MARS 1969 SYSTEM

Diameter of Antenna (ft)	Required D^2W Corrected for Constant Pointing Error of $\pm 1^\circ$ $Ft.^2W$	RF Power (Watts)	Efficiency of Klystron (%)	Overall Efficiency (%)	Input Power (Watts)	Power Supply Weight (lbs)	Antenna Weight (lbs)	System Weight (lbs)
9	4260	52.6	29.2	21.9	240	90.2	31.6	121.8
10	4490	44.9	27.2	20.4	220	82.7	39.0	121.7
11	4800	39.7	25.7	19.3	206	77.5	47.1	124.6
12	5160	35.8	24.2	18.1	197	74.0	56.1	130.1

The results show that the optimum system would require 50.0 watts at 9.6 foot diameter. It was rounded off at 10 feet and 50 watts to remain within the optimum operating range of a klystron and to maintain a higher power for emergency transmission through an omnidirection antenna.

C. Communication System Capability

Figure 3.3.2-1 shows the nominal data rates declining with respect to days after encounter for the last day in the launch window for the Mars 1969 mission. The required data rate for end of the mapping period for other than nominal orbits is shown on the short line that, of course, crosses the data rate capability curve on the 26th day for the nominal orbit and nominal capability of the communication system.

It can be seen that anytime before the end of the nominal mapping period the excess data rate is adequate for the data requirements of the diagnostic and other scientific instruments. After that date the data rate is no longer high enough to keep up the same rate of picture acquisition, but it is not required because the mapping mission is over and pictures can be transmitted in accordance with the prevailing data rate. Color and high resolution photographic coverage can be increased by continuing to operate these I.O.'s on a continuous basis after completion of the mapping mission. In addition, areas previously identified in the stereo map as having interesting features can then be photographed by the medium resolution color and high resolution camera set as they become available to the orbiter by commanding the appropriate timing and PHP pointing commands (if the selected areas are not directly on the orbit track).

D. Effect of Orbit Variation on Communication System Operation

In the case of an orbit with less than the period of the nominal 1 x 19 orbit, the communication system will not be able to transmit all the pictures obtained during the TV portion of the orbit. Since lower priority can be assigned to the high resolution and color pictures, the spacing of the stations from which these pictures are taken can be increased in order to accommodate the lower orbit. In the case of an orbit with a longer than nominal period where there is a higher ratio of communication time to TV acquisition time, the ratio of image orthicon colored and high resolution frames to vidicon stereo frames can be increased. Modification of the television operating sequence is accomplished after several days of orbiting during which time the parameters of the orbit are determined. Before that time, the nominal sequence is pre-programmed and initiated by light sensors which indicate when the orbiter has crossed the terminator as described in paragraph 3.3.3-E.

E. Effect of Variation in Communication Quality on Communication System Operation

One of the advantages of the long periods of continuous communication from the orbiter to earth listening stations is the opportunity to monitor the "hearing" conditions on the particular transmission period. If hearing conditions are favorable, the orbiter can be commanded to step up the data transmission rate. Because each TPR has a capacity of 1×10^9 bits of data, which is approximately the planned data requirement per orbit, the excess storage capacity of the second TPR can be used to supply information to be transmitted during the favorable periods of hearing conditions, thus taking full advantage of the communication systems up ability during these improved conditions. The two TPR's can be filled during the first mapping pass by increasing the number of color and high resolution picture sets acquired during the longer station intervals from station 19 out to the terminator, or out to the point where the surface light level photo cell monitor overrides the programmed picture exposing commands. The second TPR is also useful to enable the vehicle to accept and record information relayed from the lander to the orbiter if a relay opportunity occurs during an earth communication period while the first TPR is supplying data to the transmitter.

3.3.4 LANDER COMMUNICATION SYSTEM

A. Lander-Orbiter Relay Link

The telemetry link from the Landers to Earth uses the Orbiter as a communications relay in order to take advantage of the high data rate of the Orbiter communication system. The Lander VHF transmitter power level is limited to 25 watts by the maximum permitted by solid state devices. It transmits through an omni-directional antenna without the reliability and weight penalty that would be incurred by tracking the Orbiter with a directional antenna of higher gain.

The data rate of a relay link utilizing the omnidirectional antennas in both the Lander and the Orbiter is quite low even at the close distances obtained for a short time when the Orbiter is in line-of-sight with the Lander near perifocus.

Since the Planet Horizontal Package mounted on the Orbiter is oriented to the local vertical, it offered a ready-made opportunity to mount a narrow beam antenna on this PHP which would automatically be oriented to the planet center. The frequency utilized in the VHF relay link dictated the choice of a yagi antenna. The beamwidth of this yagi antenna is designed to contain the planet of Mars from an altitude of approximately 2,300 n.mi. and has a gain of 10 db.

There is a slight drop in gain if the Lander is on the horizon of the Orbiter when the Orbiter is at altitude of less than 2,300 n.mi., for in this situation, the Lander is outside the 3 db beamwidth of the antenna. But this phenomena exists for only a very few minutes near perifocus and operation of the relay link is not significantly degraded.

B. Orbiter/Lander Line-of-Sight

The operation of the orbiter/lander relay link is also affected by the ratio of orbital periods to the rotational period of Mars. The orbiter to lander line-of-sight time and range data for the two landing sites specified by Mission Analysis for the Mars 1969 mission were studied with the aid of a computer program. Plots of range versus time are given in Figures 3.3.4-1 through 3.3.4-5. The governing considerations and assumptions for this data are as follows: (1) velocity increment at orbit injection is applied impulsively at the perifocus of the approach hyperbola, (2) approach geometry for the trajectory on the first day of the launch window on January 9, 1969, was used, (3) spherical planet is assumed, (4) orbits studied were the nominal 1,000 n.mi. periapsis altitude x 19,000 n.mi. apoapsis altitude, hereafter referred to as a 1 x 19 orbit, the two limiting cases 1 x 14.5 and 1 x 25, the 1 x 17.5 orbit which has a period of close to the period of rotation of Mars, and the 1 x 22 orbit with a nominal spacing between the 1 x 19 and the 1 x 25 orbits, (5) zero time is the time of injection of the orbiter into orbit, (6) orbit planes are all inclined 55^{o} with respect to Mars equator, (7) the 24^{o} latitude landing site is in the plane of the orbit at the instant when the line-of-sight from orbiter to this landing site is lost,

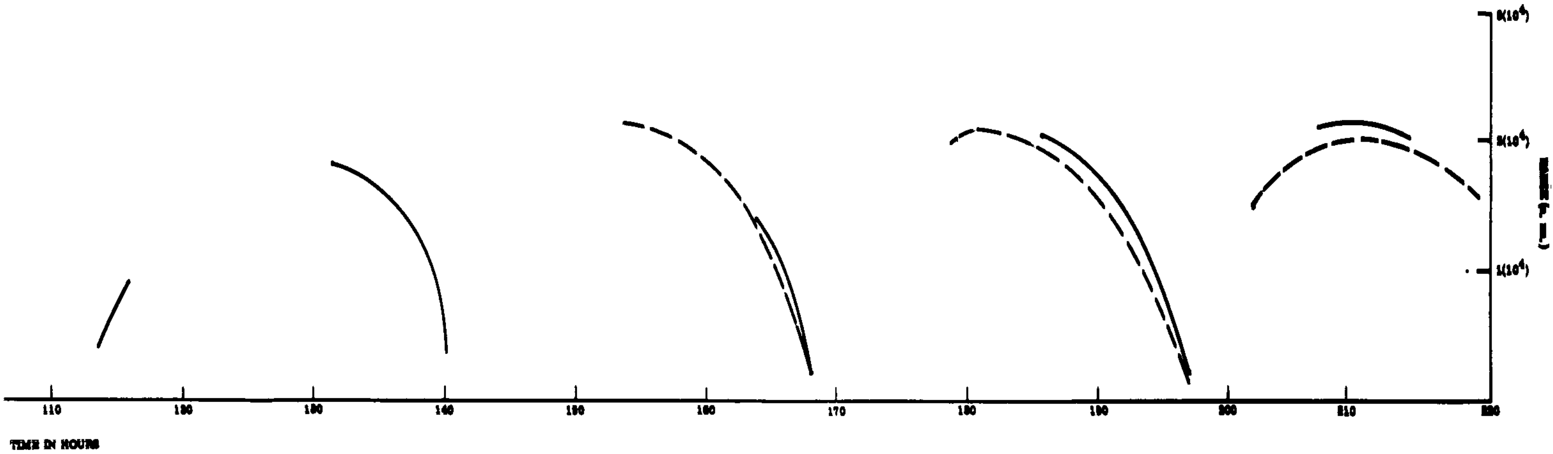


Figure 3.3.4-1. Lander-Orbiter Line-of-Sight 1 x 19 (1000 nmi) Orbit Mars

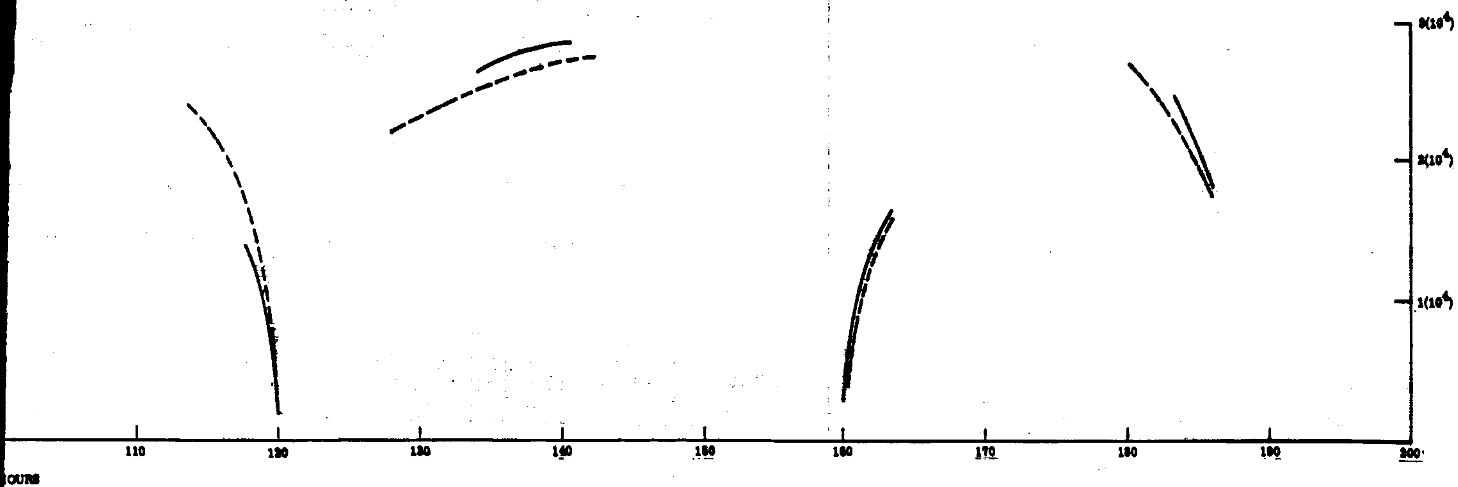


Figure 3.3.4-3. **Lander-Orbiter Line-of-Sight 1 x 25 (1000 nmi) Orbit Mars**

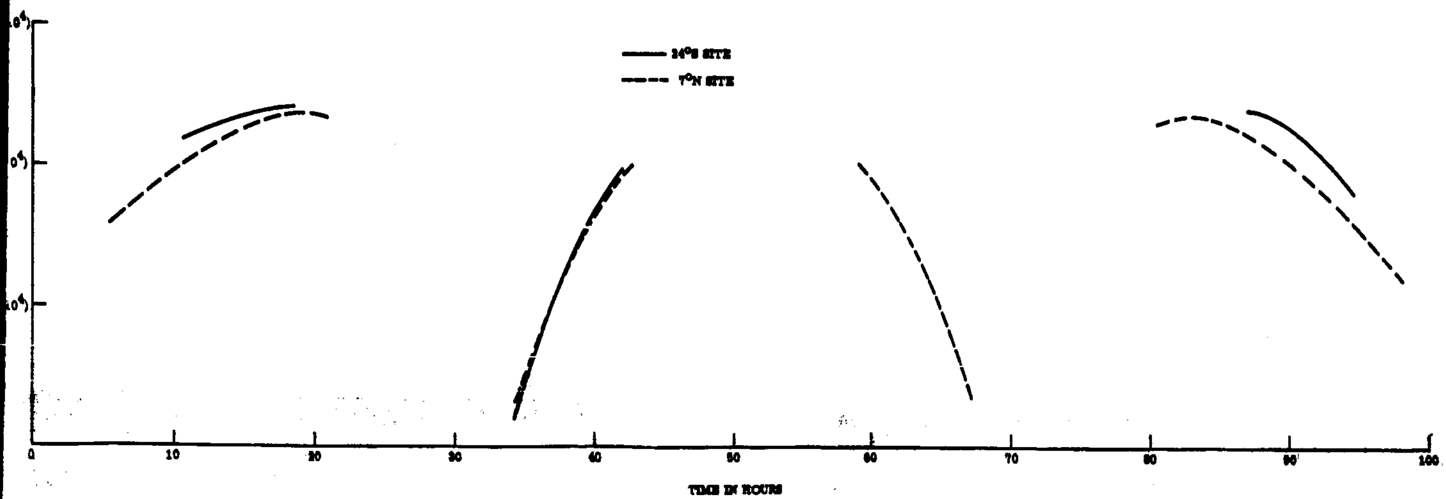
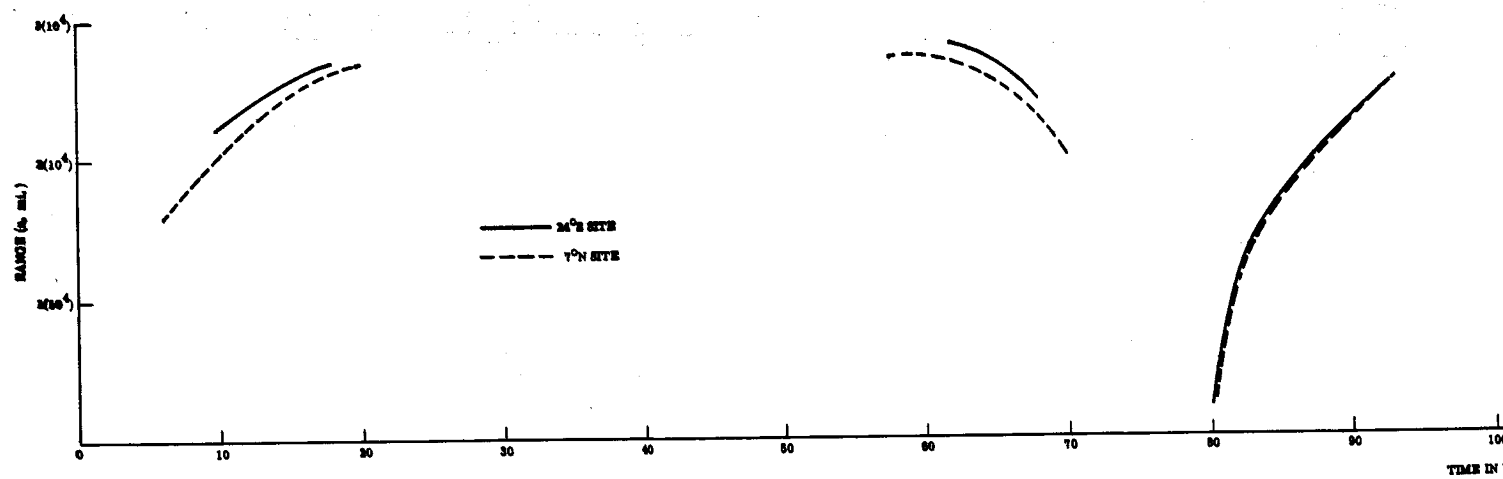


Figure 3.3.4-5. Lander-Orbiter Line-of-Sight 1 x 22 (1000 nmi) Orbit Mars



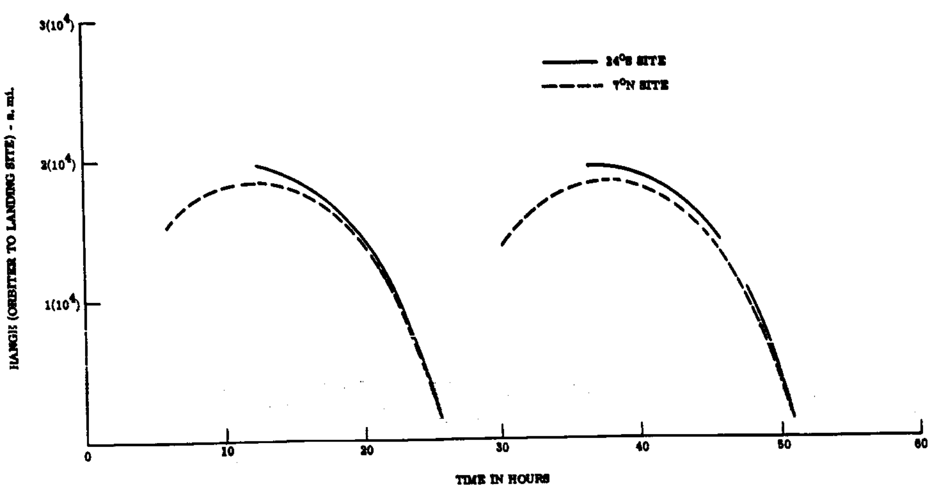


Figure 3.3.4-4. Lander-Orbiter Line-of-Sight
1 x 17 (1000 nmi) Orbit Mars

(8) line-of-sight between the orbiter and the lander sight is lost when the orbiter passes below a plane tangent to the landing site. Intervals where curves do not exist in the figures correspond to periods of time during which the orbiter is below the landing site horizon, (9) the curves were extended to a number of orbits which correspond to a period of time during which Mars rotates approximately an integral number of times on its polar axis, thus providing patterns that tended to repeat themselves periodically. It can be seen in Figure 3.3.4-4 (1 x 17.5 orbit), that the line-of-sight to the Lander changes very slowly because this is nearly a synchronous orbit; and therefore, we have for a time, an opportunity for the operation of a relay link at short range once per orbit.

Since the orbit period cannot be permitted to remain exactly synchronous with Mars' rotational period, because of the resulting limitation on the TV mapping mission, the perifocus will drift away the Lander site and line-of-sight opportunities will occur at continually increasing ranges.

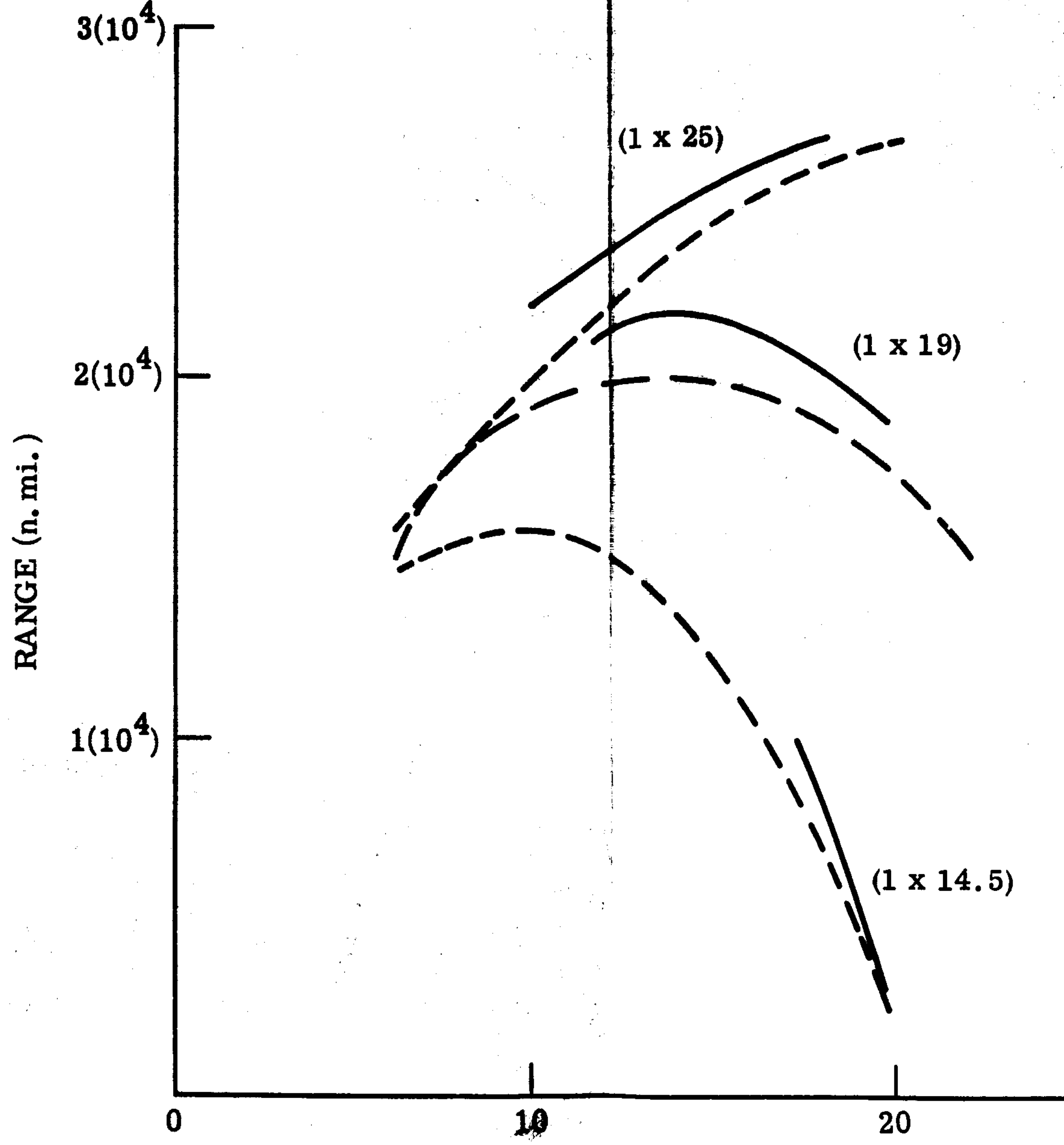
Synchronous orbits will not cause extended blackout of the relay link line-of-sight opportunities because of the highly eccentric and inclined orbit. The Lander sites will come into view of the Orbiter sometime during the apofocus region of the orbit.

C. Relay Link Operation

In Figure 3.3.4-6, the line-of-sight range and times are indicated on one graph for both Lander sites and for the minimum, nominal, and maximum expected apoapsis altitude for the first   hours of the mission. These curves indicate that it is possible to pick out a preprogrammed time for the first relay communication from each Lander to the Orbiter. A few days after encounter, when the orbit has been carefully determined by tracking from earth, specific times for the occurrence of relay link operation can be computed by a program like the one used for this study and transmitted to the orbiter to make maximum use of the relay link. The permissible data rate between the Lander and the Orbiter is a function of the square of the range between them. Therefore, the relay link achieves maximum data transmission when short high power periods of communication are used when Orbiter and Lander are closest. The power supply and communication system on board the Lander are planned to take advantage of this situation. The definition of the orbit, of course, allows planning the Lander operation to insure a 100% charge available in the secondary battery on board the Lander when the relay period is expected to occur. Data rate estimates for the Lander to Orbiter relay link are based on utilization of minimum range line-of-sight occurrences.

When the orbiter is approaching a landing site horizon, it transmits a command continuously, and this signal when received by the Lander VHF receiver is used to trigger maximum data rate relay transmission back to the Orbiter. Orbiter transmitted command will be the normal mode of initiating Lander VHF transmission because of the assurance that the Orbiter is in line-of-sight with the Lander.

The information capacity of the Lander/Orbiter relay link is a function of the available power and communication range. The Lander power supply will provide energy from its RTG and secondary battery for 38 minutes of continuous VHF transmitter operation and will then require approximately 9 hours to recharge the battery. The nominal available data rate was calculated based on the line-of-sight opportunities and ranges for the first 196 hours of Lander life on the basis of the selected 1 x 19 (1,000 n.m.) orbit, for the 7 N. latitude Lander site. See Figure No. 3.3.4-1.


3(10⁴)
2(10⁴)
1(10⁴)
RANGE (n. mi.)
(1 x 25)
(1 x 19)
(1 x 14.5)
0
10
20

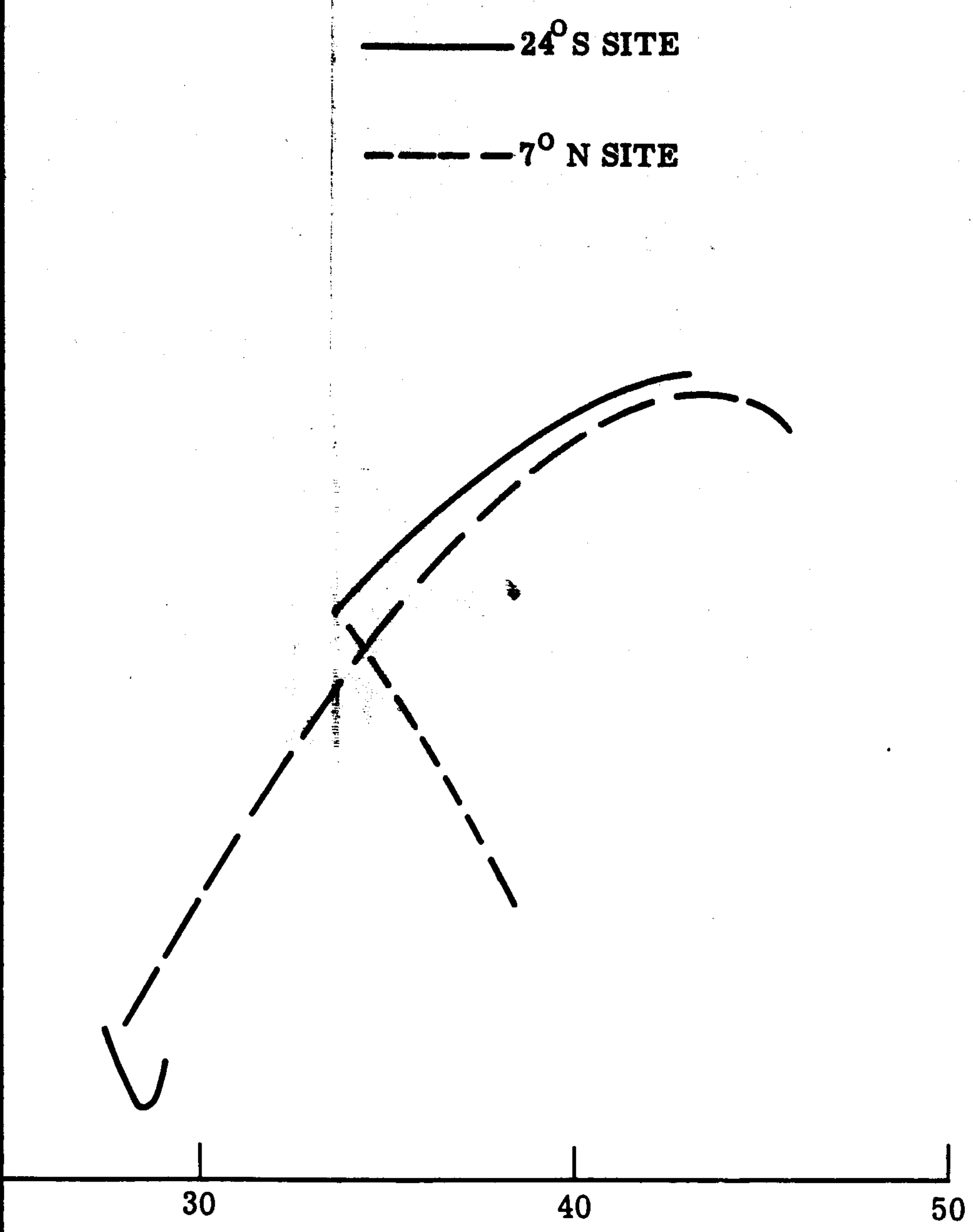


Figure 3.3.4-6. Line of Sight Range and Times

The information transmitted in each opportunity is listed in Table 3.3.4-1 in terms of Lander television frames containing 4.1 x 10^5 data bits. Scientific and engineering information is transmitted in place of one (1) or two (2) TV frames per opportunity.

TABLE 3.3.4-1. INFORMATION TRANSMITTED IN EACH OPPORTUNITY

Hours After Encounter	Frames Transmitted During Opportunity	Transmitted Accumulations
28	125	125
58	125	250
85	113	363
139	35	398
168	35	433
196	75	508

D. Lander-Orbiter Relay During Separate Cruise and Entry

VHF omnidirectional antennas are provided on the Orbiter and Lander so that diagnostic telemetry can be transmitted from the Lander to the Orbiter after Lander separation.

Line-of-sight during Lander atmospheric entry and descent phase is assured by adding a tangential component to the velocity increment impacted to the Lander after separation to change its trajectory from that of the Orbiter, which will pass Mars at the periapsis altitude, to the impact trajectory which will put the Lander in the desired Lander site. This causes the Lander to arrive at the planet before the Orbiter passes over the landing site horizon. The descent radar incorporated in the Mars Landers is used to insure that deployment of the final descent parachute is delayed until Lander descends to an altitude of 30,000 ft., thus minimizing descent time and the corresponding required velocity increment. Although this reduces the time available for atmospheric measurements during descent, it is felt that atmospheric data should have been obtained by Mariner B capsules by the time of this mission. Thus, long descent times are not a Voyager requirement. With this short descent time of approximately 11 minutes, the Orbiter will be in line-of-sight with the Landers until the Landers have arrived on the surface of Mars. See Section 4 in the Guidance and Control Section of the Subsystem Design Volume III.

E. Lander Direct Link

The relay link is the prime mode of communication from Lander to earth during the planned lifetime of the Orbiter of 90 days. However, after the cessation of Orbiter operation, the Lander, which is in a benign atmosphere and environment and has a continuous power supply from the radioisotope thermoelectric generator can continue to operate. Therefore, it is equipped with a direct communication link to Earth. This secondary link can be a back-up mode in the event of failure of the Orbiter communication system, or in case of failure of the Orbiter to achieve capture. Since the scientific value of the second 90 days of Lander operation is much less than the previously acquired information and is confined to observation of seasonal and long term time changes, the limited data rate capability of this secondary communication mode is quite adequate in Lander direct link.

An S-Band transmitter is provided in the Lander. It transmits through a helix antenna which can be oriented toward Earth. Section 4 of Volume No. III discusses this antenna pointing mechanism. Power is available for two (2) transmission periods of 30 minutes, each within the time during each Martian day that Earth is over the Lander site horizon. The information capacity of this link is 235 Lander TV frames in the second three months of Lander operation. The same klystron power amplifier that will be developed for use in the orbiter is operated in the Lander direct mode at a power level of 70 watts. This high level is within the range of a klystron, and it insures some communication ability through the omni antenna in the event of failure of the earth oriented high gain antenna.

Even though the design life of the Martian Landers is six months, it is not planned that the operation of the Lander will be deliberately restricted to that length of time. If the Lander continues to function then there is no reason that data should not continue to be obtained although at a much lower rate because of the continually increasing range from Mars to Earth.

3.3.5 SEQUENCE OF EVENTS

A. Spacecraft

With these considerations and the enlarged definition of the missions, a sequence of events was compiled Ref Vol IV Section 2.2 that listed the required maneuvers and events in the transit and landing portions of the mission. The spacecraft sequence of events covers the mission from trajectory injection to orbit insertion.

B. Alternate Operating Modes

The sequence of events listed above is based upon planned operating modes with all systems operating as designed. In order to increase the probability of mission success with a minimum loss of capability, alternate modes of operation are provided to compensate for selected component failures. If it is discovered on the beginning of the transit mode that the Canopus tracker is inoperative, the antenna can be commanded to assume appropriate correct hinge angles based on the knowledge of the position of the spacecraft in its transit trajectory. The Earth sensor mounted coaxially with the high gain antenna can substitute for the Canopus tracker. This requires that the hinge angles of the high gain antenna be updated from time to time as the spacecraft proceeds on the trajectory towards Mars. If the body mounted image orthicon camera that is to be used to acquire the planet based information for the terminal guidance observation fails to function, the PHP could be unhinged by command and one of the Image Orthicon television cameras in the PHP could be substituted, or the navigation of the spacecraft can be accomplished by Earth based information obtained from the two-way doppler tracking of the spacecraft. Degrees of redundancy and alternate modes in the various subsystems are described in the pertinent sections of the subsystem design volume.

C. Power Profile Orbiter

Power requirement estimates for various instruments, sensors, and components aboard the vehicle were obtained from the subsystem design groups and, with the sequence of events, a power profile was compiled for the orbiter. The power profile mode numbers and component groups are shown on the matrix Figure 3.3.5-1. Mode power totals are indicated on the power profile Figure 3.3.5-2. Vehicle characteristics such as thermal inertia were used to conserve power during communication periods which, of course, require the most power on the orbiter during transit.

Maneuvers requiring changes in attitude of the spacecraft were arranged so that the departure from the sun orientation would be the last rotation performed before maneuver. The first vehicle attitude change after completion of a firing of a correction velocity impulse would be towards the sun attitude to regain the use of the solar array.

There is ample time during transit to recharge the secondary batteries following a maneuver. After separation of the second Lander, the communication to Earth is limited to insure that the vehicle batteries are fully charged during the 50 minutes the vehicle is in the orbit insertion attitude. Once the Orbiter is in orbit and acquires the cruise attitude, batteries are not utilized until it enters Mars' shadow. The enclosed power profile for the Mars 1969 Orbiter (Figure 3.3.5-2) does not show that communication to Earth can be maintained while a relay broadcast from the Lander is being recorded on-board the Orbiter. However, such communication which would demand peak power of 472 watts, 32 watts above the rated capacity of the solar array, could be handled by utilizing the secondary batteries to supply the 32 watts power deficit.

D. Orbiter Power Supply

The Power Supply Subsystem was designed on the basis of the power profile discussed above. A Solar Cell Array and secondary batteries are used in Mars 1969 Radioisotope thermoelectric generators are quite competitive in weight, but were not selected because of limited availability of radioisotope fuel. See Section 6, Power Supply of Volume No. III for a more complete discussion.

The selected orbit will have no shadow time until 83 days after encounter in the worst case; therefore, no part of the solar array area is allocated to recharging batteries during the orbiting mission. See Paragraph No. 3.3.5F for a discussion of sun occultation.

E. Back-Up Modes of Power Supply During Transit

If the attitude control system fails so that the vehicle attitude is uncontrolled and energy from the solar array is not available, the secondary batteries on board the Orbiter and on board the two Landers, as well as the excess power from the Lander radioisotope thermoelectric generators can be used to power the Orbiter's communications system through the omni antenna to transmit diagnostic information to Earth for diagnosis, analysis and possible solution.

F. Sun Occultation in Orbit

(1) <u>Orbit Geometry</u>

The precise definition of the desired capture orbit is a function of operational considerations, arrival hyperbolic geometry of the trajectory and of course a required periapsis impulsive retro thrusting. This problem was studied as a two-body problem

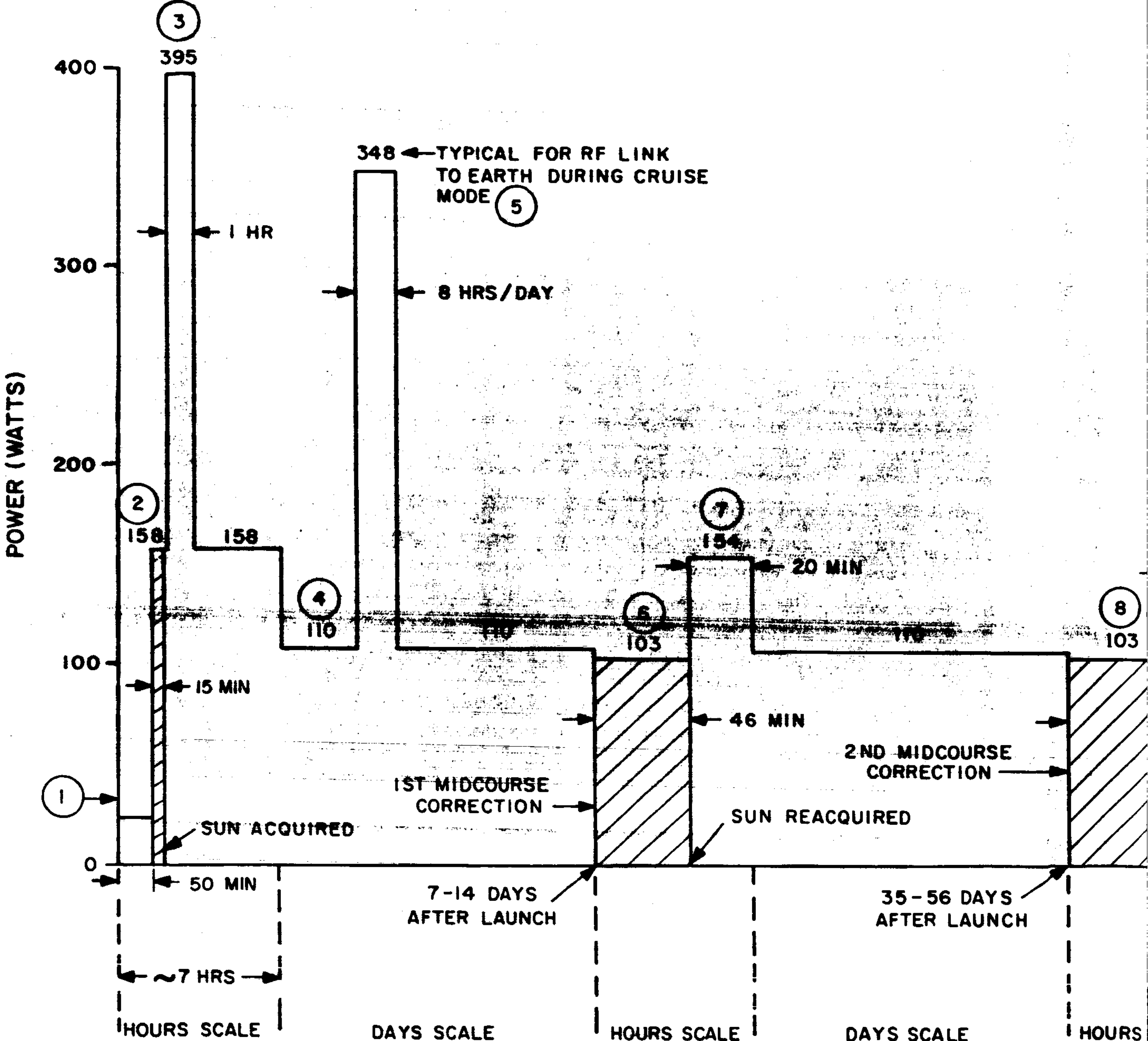
FUNCTION	1 Launch & Injection	2 Initial Orient.	3 Initial Orient. & Earth Link to Verify Canopus	4 Cruise	5 Cruise & Earth Link
I. COMMUNICATIONS					
• Earth Link			232		
• Lander Link (Relay)					
• Data Storage & Processing		4	4	4	4
• Thermo Plastic Tape Rec.					
• T.V. Cameras & Elect.					
• Commands & Tracking		8	9	8	9
• Power Conversion & Control			5		5
II. GUIDANCE & CONTROL					
• Att. Control (Less Gyro)		65	65	28	28
• Gyros & Elect.	16.5	16.5	16.5		
• Hi Gain Ant. Pointing		29	29		
• P.H.P. Pointing					
• Body Mounted I.O. Camera & Elect.					
III. THERMAL CONTROL		25	25	50	50
IV. SCIENCE				10	10
V. ENG. DATA	4	10	10	10	10
TOTALS	~20	157.5	395.5	110	348

NOTE: Numbers in M

ORBITER															
	EVENT														
	6	7	8	9	10	11	12	13	14	15	16	17	18	19	20
											Orbiter Over Dark Mars			Orbiter Over Sunlit Mars	
	1st Midcourse	Canopus Reorient	2nd Midcourse	Cruise + Hi Gain Point	Term Guid. Obser. + Earth Link	Final Traj. Corr.	Cruise + Term. Guid. Observ.	1st Lander Ejection	2nd Lander Ejection	Orbit Injection + Lander Link	Orbiter Science	Orbiter - Lander Relay Mode	Orbiter Earth Link	TV. Lander Relay	Orbiter TV
					232								232		
										18		18		18	
	4	4	4	4	4	4	4	4	4	4	4	4	4	4	4
					25		25				25	25	25	50	50
														140	140
	8	8	8	8	9	8	8	8	8	9	9	9	9	9	9
					5					1	1	1	5	5	5
	65	65	65	28	28	65	28	65	65	65	28	28	28	28	28
	16.5	16.5	16.5			16.5		16.5	16.5	16.5					
				29	29		29				29	29	29	29	29
											41	41	41	41	41
					25		25								
		50		50	50						25	25	25	25	25
				10	10		10				25	25	25	25	25
	10	10	10	10	10	10	10	10	10	10	10	10	10	10	10
	103.5	153.5	103.5	139.0	427.0	103.5	189.0	103.5	103.5	123.5	197	215	433	384	366

atrix Denote Power Demands in Watts

Figure 3.3.5-1. Orbiter Power Profile Mode Numbers Matrix



POWER (WATTS)
400
300
200
100
0
3
395
1 HR
348
TYPICAL FOR RF LINK TO EARTH DURING CRUISE MODE
5
8 HRS/DAY
2
158
158
4
110
110
6
103
7
154
20 MIN
110
8
103
15 MIN
46 MIN
2ND MIDCOURSE CORRECTION
1
1ST MIDCOURSE CORRECTION
SUN REACQUIRED
SUN ACQUIRED
50 MIN
7-14 DAYS AFTER LAUNCH
35-56 DAYS AFTER LAUNCH
~7 HRS
HOURS SCALE
DAYS SCALE
HOURS SCALE
DAYS SCALE
HOURS

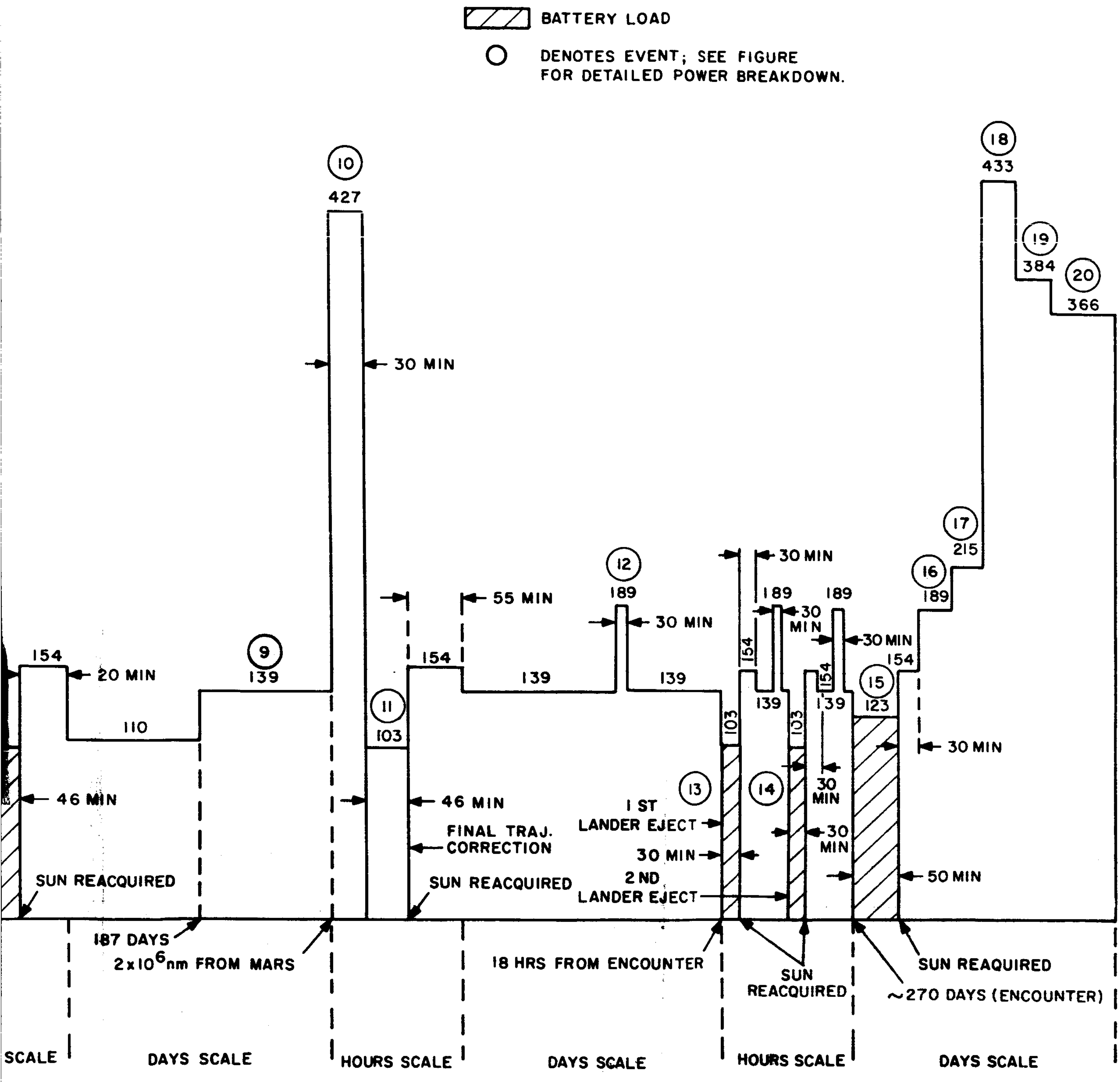


Figure 3.3.5-2. Orbiter Power Profile

involving the target planet and the vehicle. The desired 55° inclination of the orbit plane to Mars equator, and the hyperbolic excess velocity for a particular day or trip in the launch window established the displacement of the hyperbolic approach vector from the center of the planet and defined the orbit in its plane. Figure 3.3.5-3 shows the arrival geometry with the hyperbolic excess velocity vector aiming at the center of the planet. Figure 3.3.5-4 shows the required displacement B of the hyperbolic excess velocity vector, omega is the angle between the line parallel to that vector at the center of the planet to the radius vector of periapsis and this is in the orbit plane. Figure 3.3.5-5 shows the orbit plane orientation. Figure 3.3.5-6, orbit geometry, show an angle beta between OS, the radius vector and direction of the sun line projection and OP, the radius vector at periapsis. Results of this analysis for the first day launch, 10 January, 1969, and the last day of the launch window, 9 February, 1969, show that this angle beta at encounter is 27.38° and is minus 14.86 degrees on the last day of the launch window at encounter. The displacement of the orbit around the planet Mars is effected by Mars' seasonal progression and also is a function of the planet's oblateness. Seasonal progression for the earliest launch date is -.625° a day, and for the latest launch date -.604° per day. Coefficient J for Mars oblateness was estimated to be 2.58×10^{-3}. The daily change in periapsis angle due to oblateness is 0.47° per day. The daily change in "longitude of ascending node" due to oblateness is -.084 degrees per day.

(2) Duration of Orbiter Mission

This study covered the movement of the orbit for the 90-day duration of the orbiting mission. Orbiter life was limited because with an all gas attitude control system controlling the attitude of the orbiter, there is a daily consumption of cold gas stored in the vehicle due to unbalanced solar torques and the gravity gradient history. Once a complete map of the planet is obtained, and a reasonable number of medium and high resolution pictures of smaller areas of the planet are obtained, the mission value to be gained by extending the life of the orbiter grows rapidly smaller and smaller, approaching zero asymptotically. It then becomes more worthwhile to add more instruments that operate in the early portion of the orbiting life rather than to extend the life past the point of adequate return for the investment of attitude control gas weight. Of course, since the orbiter has a high data rate capability, it is desirable to maintain the orbiter operating as long as possible to be available for use as a communications relay station transmitting high volume television data from the lander to earth. The capacity of this relay link is approximately 5,000 television frames in the 90 day relay mode period. Since the lander is also equipped with a direct mode of communication to earth which has a capacity in second 90 days of approximately 235 frames, it was decided that enough television information would have been transmitted from the lander in the 90 day orbiter life. Therefore, the 90 day life of the orbiter was deemed adequate.

(3) Effect of Mission Duration on Orbit Geometry

The angle beta between the sun line and periapsis for the first case will now have proceeded to -39.05 degrees at the end of the 90 day period and for the last day of the launch window would have moved from -14.86 degrees to -75.52 degrees. Thus, it is seen that an assumption of periapsis location on the noon meridian at some time in the launch window is correct.

This investigation was used to determine the angle between the sun line and the orbit plane. Since the planet horizontal package is aligned perpendicular to the orbit plane and is rotated around an axis parallel to the orbit plane as the orbiter proceeds around

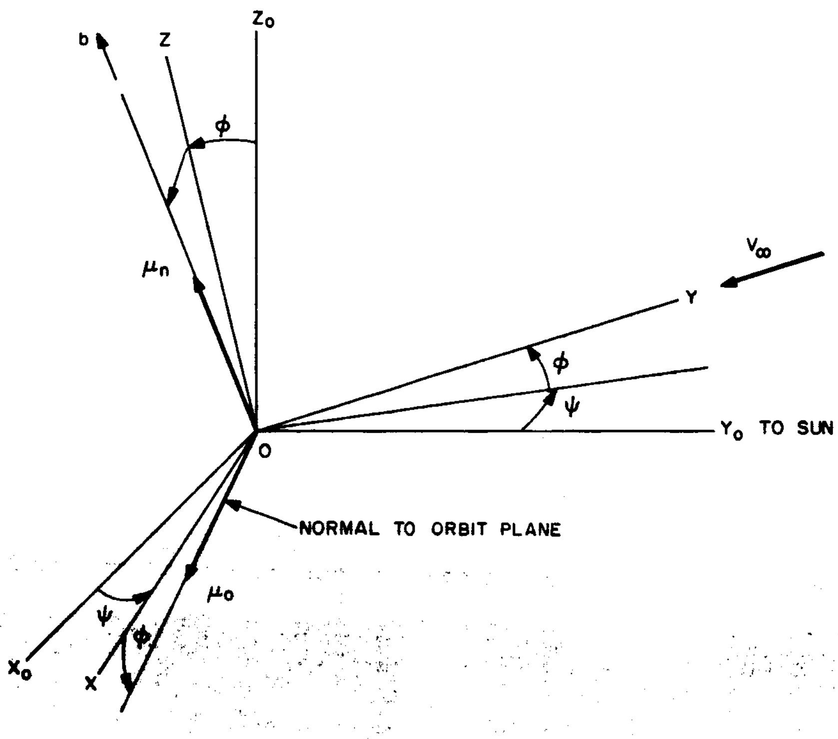


Figure 3.3.5-3. Coordinate Systems

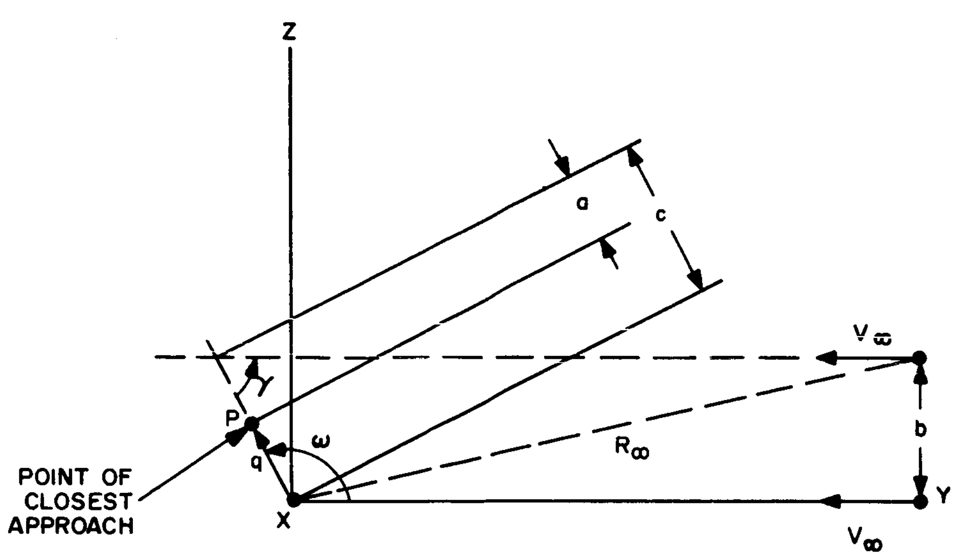


Figure 3.3.5-4. Motion in the Y-Z Plane

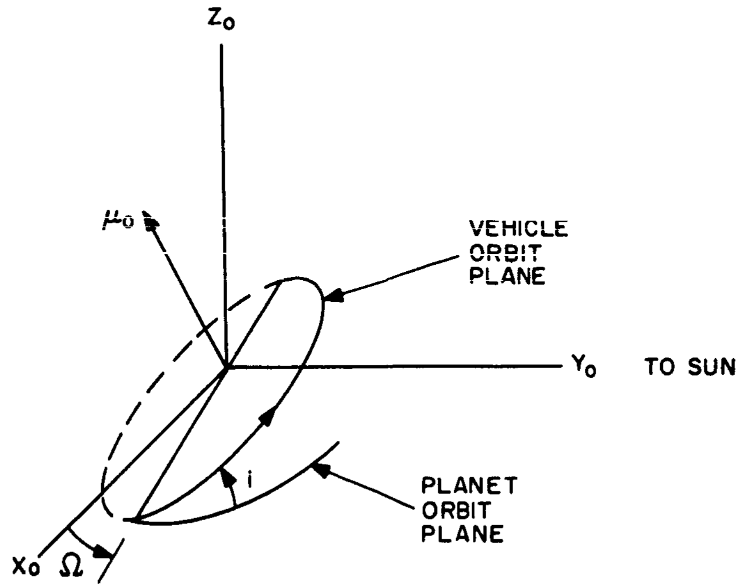


Figure 3.3.5-5. Orbit Plane Orientation

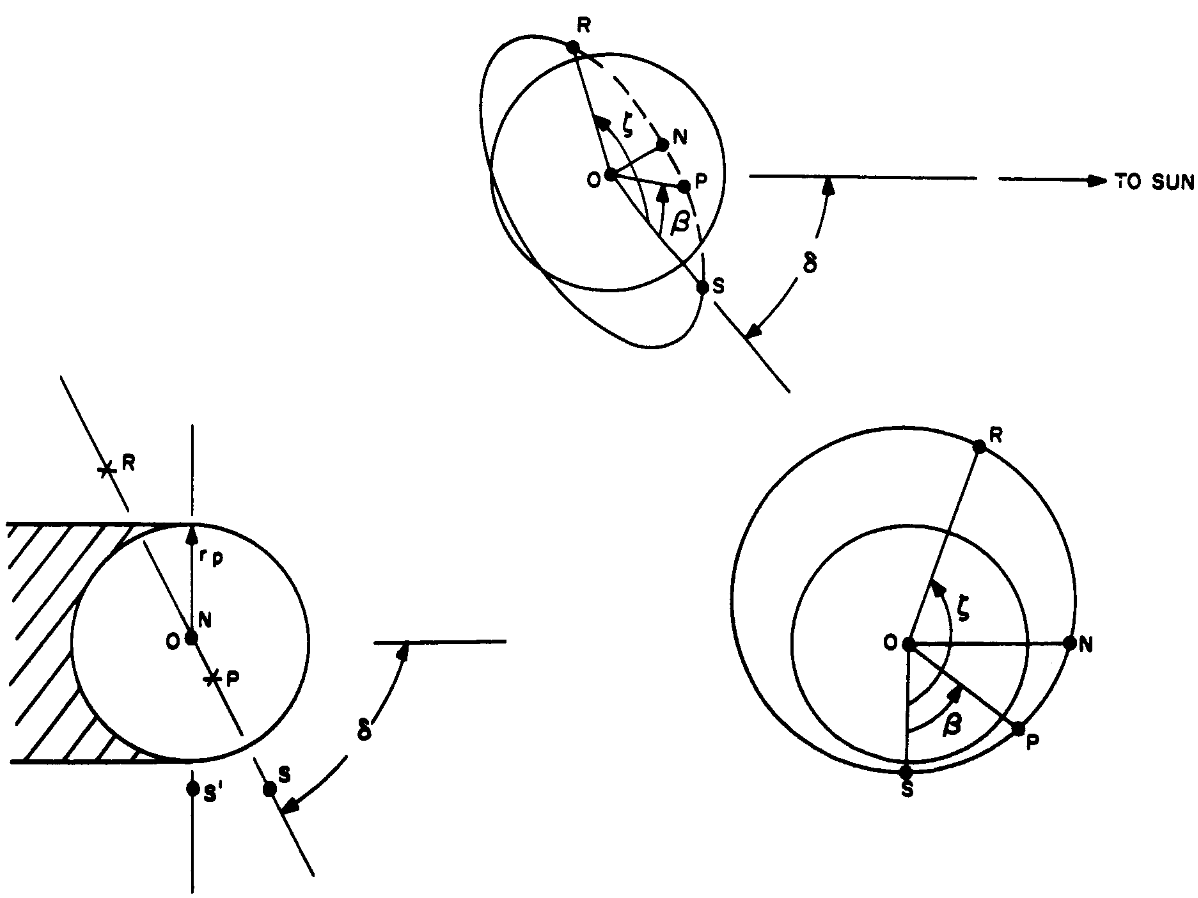


Figure 3.3.5-6. Orbit Geometry

the orbit, this angle determines the thermal environment of the planet horizontal package. This angle is shown in the orbit geometry figure No. 4 as the angle delta (δ). For the first day of the launch window sigma is -25.20°, at the end of 90 days it is -25.14°, and for the last day of the launch δ at encounter is -30.66°. At the end of 90 days it has changed to -14.51°. The thermal environment for the planet horizontal package was studied in detail for an angle of -25°.

(4) Sun Occultation

This analysis was used to determine the occurrence of shadow time in the orbit and the duration of this shadow when it does occur. For the chosen orbit it was determined that in the case of the first day of the launch window the orbiter would be continuously sunlit for its entire operational lifetime of 90 days. For the last day of the launch window it would be in shadow only at the 83rd day. At 90 days after encounter, this shadow would have increased to 36.2 minutes. The maximum shadow occurs 112 days after encounter and is approximately 53 minutes in duration.

(5) Post 90-Day Orbiter Operation

Since it is difficult to predict the precise quantity of attitude control gas that must be carried to satisfy the requirements of the 90 day stay in orbit, and if the vehicle continues to function as a communications relay station and as a television mapping orbiter, it was felt that operation would not cease but would continue until either some serious component failure occurred or the vehicle exhausted its supply of attitude control gas.

Since solar cells were the prime candidate power source, it was recognized that rechargeable batteries would have to be provided in the vehicle to supply electrical power when the vehicle had to be oriented away from the sun during maneuvers such as midcourse or terminal guidance correction, lander separation, and orbit insertion. Of course for terminal and midcourse corrections the landers are still attached to the orbiter, and their power supplies and secondary batteries could be used to augment the power supply on board the orbiter. However, the orbiter must supply its own stored electrical energy during the orbit insertion maneuver which occurs after the separation of the two landers. The length of time for this maneuver is estimated to be approximately 50 minutes which is very close to the maximum shadow time expected in the post 90 day operation of the orbiter. Therefore, the battery that is adequate for the orbit insertion maneuver would also enable the vehicle to pass through the expected maximum shadow time without exhausting its stored energy. No solar cells would be provided to recharge this battery during orbital operations because of the extended period when no shadow time occurs. To obtain energy from the solar array to charge the batteries when shadow is encountered, the communication time can be reduced because data requirements are much lower at this time in the mission.

(6) Sample Calculations of Sun Occulation Orbit

Inputs:

From Trajectory: V_{∞}, ψ, φ, Ω.

From Parent Planet Characteristics: e, GM, rp, $\dot{\Omega}$.

From Operational Restrictions: i, hp, ha, T, hemisphere.

From Parent Planet Characteristics and Operational Restrictions: $\dot{\theta}_p$, $\dot{\lambda}$

The undefined quantities are:

- T; The time in days after initial capture at which shadow is also to be investigated.
- $\dot{\Omega}$; The average seasonal change of the planet sun line over the interval T.
- $\dot{\theta}_p$; The daily change in periapsis angle due to oblateness.
- $\dot{\lambda}$; The daily change in "longitude of ascending node" due to oblateness.

Outputs:

Trajectory Parameters: θ, ω, e_N

Initial Capture Orbit Definition: λ, θ_p, e_E

Initial Orbit to Planet-Sun Line Geometry: β, δ

Final Orbit to Planet-Sun Line Geometry: β_F, δ_F

A considered recommendation for Mars trajectories for 1969 was a launch window of forty days from 10 January to 9 February with trip times of 270 days to 280 days respectively resulting in arrivals of 7 October and 16 November. Launches performed within this launch window are assumed to be represented by the two extremes.

Launch Trip Time Arrival Date	10 Jan., 1969 270 Days 7 Oct., 1969 3.7 km/sec.	9 Feb., 1969 280 Days 16 Nov., 1969 4.2 km/sec.
V_∞	-16°	-19°
δ_p	95°	60°
ζ_p		

	Case I Earliest Launch Date	Case II Latest Launch Date
V_∞	1.212×10^4 ft./sec.	1.380 no^4 ft./sec.
ψ	275°	240°
φ	16°	19°
Ω	302.3°	277.0°
e	241.2°	24.2°
GM	$.1505 \times 10^{16}$ ft.^{3}/sec.^{2}	$.1505 \times 10^{16}$ ft.^{3}/sec.^{2}
r_p	1800 n.m.	1800 n.m.
$\dot{\Omega}$	$-.625^\circ$/day	$-.604^\circ$/day

The results are:

	Case I		Case II
θ	-26.68^{o}	θ	-24.716^{o}
ω	112.07^{o}	ω	108.48^{o}
e_H	2.6616	e_H	3.1542
λ	180.90^{o}	λ	178.92^{o}
θ_p	260.23^{o}	θ_p	248.15^{o}
e_E	.76271	e_E	.76271
β	27.38^{o}	β	-14.86^{o}
δ	-25.20^{o}	δ	-30.66^{o}
β_F	-39.05^{o}	β_F	-75.52^{o}
δ_F	-25.14^{o}	δ_F	-14.51^{o}

To determine the days after capture for which shadow was first encountered, the orbit was investigated for T of 80, 82, 84, 86 and 88 days. T of 80 and 82 days were without shadow. The shadow parameters for the other values of T were:

	84 days	86 days	88 days
ζ_1	$= 181.25^{o}$	$= 178.17$	$= 175.93$
ζ_2	$= 189.13^{o}$	$= 192.50$	$= 194.81$
η	= 0.008574	= 0.015095	= 0.019430
shadow	= 14.0 minutes	= 24.7 minutes	= 31.8 minutes

Shadow time is based on an orbital period of 1635 minutes. Shadow begins for this configuration at approximately 83 days.

G. Lander Sequence of Events

While the Lander is attached to the Orbiter, Lander diagnostic information will be transmitted through the hardwire connection to the Orbiter communication system for transmission to Earth. The Lander RTG power supply must, of course, be cooled; the coolant pump is powered by the RTG. All commands to be executed by the Lander at programmed times will be transmitted through the Orbiter communications and command systems into the Lander sequence timer system. After the Lander is separated and has spun up, a series of diagnositc telemetry transmissions will take place from the Lander to the Orbiter on a 71 minute cycle so that the Lander secondary battery can be brought up to a full charge between successive transmission periods.

After separation, each Lander becomes a spin stabilized space vehicle in its own right. However, there is no provision on-board the Landers to correct their attitudes if spin stabilization does not function correctly or if some orbit disturbance in the attitude is brought about by a meteorite strike. Therefore, each Lander is equipped with rate gyros and three-axis accelerometers. Just before separation, these gyros are energized and a complete history of vehicle attitude changes during the separation and subsequent spin up and firing of the solid rocket propulsion motor is obtained from the rate gyros and telemetered back to the Orbiter. After the firing, the Lander propulsion, the gyros are shut down and the three-axis spring mass accelerometers are continuously monitored in order to determine any changes in spin rate or attitude of the vehicle during the separate cruise phase after separation. Just before the expected time of arrifal at the edge of Mars' atmosphere, the communication system is turned on and the gyros are energized and the RTG cooling radiator is jettisoned. The RTG is cooled during the entry and descent phase by boiling a small quantity of water. Attitude and spin rate history are obtained through entry and descent phases. Since the Landers are equipped with descent radar, the attitude gyros are kept running through descent phase in order to provide vehicle orientation information to refine the returns from the radar systems. The three-axis accelerometers again function at impact to record the severity of the impact with the surface of the target planet. The battery capacity is sufficient to communicate to the Orbiter during the entry and descent phases.

H. Lander Power Profile

If the entry Lander survives impact with the planet, then the surface phase of Lander operation is initiated. Basic power requirement for surface operation from the Lander are listed in the power matrix (see Figure 3.3.5-7) and total 38.2 watts. This housekeeping power is a constant demand on the Lander power supply. All other power needs such as communication, television, subsurface drill operation, etc. are peaks of short duration. For example, high data transmission requiring high power transmitter take advantage of the rather short periods of time that the Orbiter is within favorable range of the Lander.

The resulting power profile is shown in Figure 3.3.5-8. Radioisotope thermoelectric generator is used for prime power supply in the Lander because of the thermal control requirements during the cool Martian nights.

I. Lander Power Supply

Because the high peak power demands can be well separated in time, the power supply of least weight is obtained when a secondary battery is incorporated in the system to supplement the output of the RTG during the short duration peak demands in the power profile. The RTG is sized so as to be able to recharge the secondary batteries during base load operation. The RTG produces 80 watts electrical gross and 72 watts, net.

The planned mode of communication from Lander to Earth is by Orbiter relay and utilizes the VHF relay link. Since Orbiter life is limited to 90 days, relay link is not available during the second 90 days of the planned Lander operation. A secondary data link is provided which transmits to Earth directly through an orientable high gain antenna with the 70 watt klystron which is the same that is utilized at the 50 watt level on board the Orbiter. The secondary battery is sized to provide power for a half hour direct link communication period. The Ni-Cad battery utilizes 29 pound and provides 157 watt hours of staved energy. The radioisotope thermoelectric generator has sufficient excess power to recharge the battery within an eight hour period. This permits two one-half hour communication periods per day.

EVENT	1	2	3	4
	Orbiter - Lander Separation	Base Load and Diagnostics	Diagnostic Communications	Entry Phase No Radar Altimeter
I. Communications				
• **Relay Transmitter (VHF)**			115	115
• **Direct Transmitter**				
• **Thermoplastic Tape Recorder**				
II. Diagnostics	10	1	1	10
III. Science				
• Experiments				26
• Panoramic TV				
• TV Microscopic Analysis				
IV. Baseload	39	39	39	39
Total Watts	49	40	155	180

Note: Numbers in Ma

5	6	7	8	9	10	11	12	13	14	15	16	17	18
Entry - Descent and Radar Alt.	Panoramic TV and Science	Science Only	Relay Link To Orbiter	TV Microscopic Biological & Petrographic Analysis	Panoramic TV and Sciences	Lander Direct Earth Link							
115			115										
						286							
	25		25	25	25	25							
10	1	1	1	1	1	1							
41	16	16		8	16	10							
	30				30								
				20									
39	39	39	39	39	39	39							
205	111	56	180	93	111	360							

ix Denote Power Demands in Watts

Figure 3.3.5-7. Lander Power Matrix

I. COMMUNICATIONS

	Power
High Voltage Power Supply (Direct Link to Earth)	280
VHF Transmitter	115
Power Amplifier (Klystron)	6
Transponder	2

Note: Other elements of communications power demand are shown in base load.

II. DIAGNOSTICS

	Power
Vehicle and Subsystem Temp.	0.125
Scientific Payload Temp.	0.100
Linear Accelerometers	0.030
Pump and Boiler Pressure	0.065
Pump Motor Speed	0.10
Pump Motor Current	0.10
Propulsion Pressure	0.010
	0.530 Watts

Rate Gyros - 9 Watts
Diagnostics (Less Rate Gyros) ~ 1 Watt
Diagnostics (With Gyros) ~10 Watts

III. SCIENCE

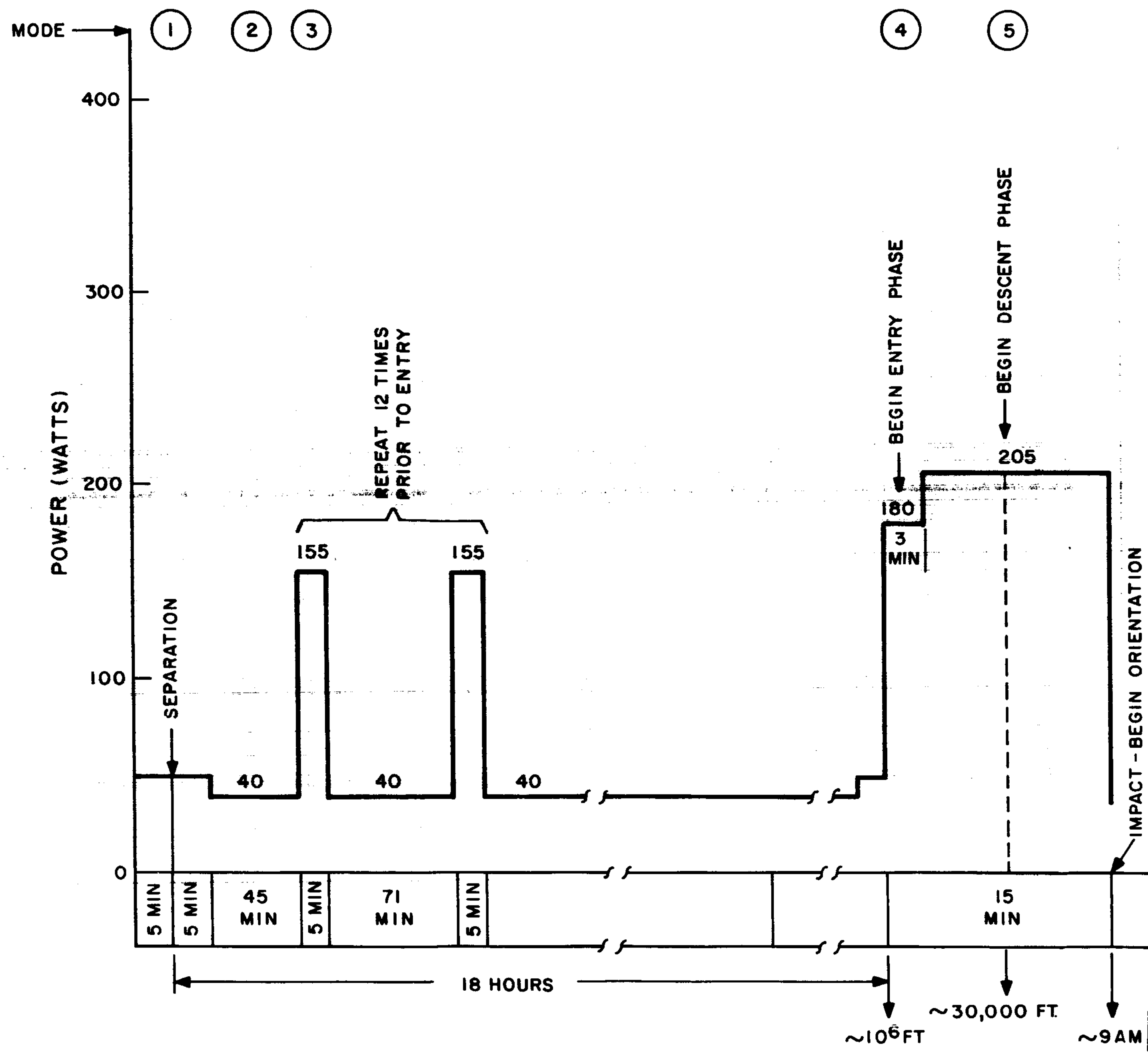
	Power
• Entry and Descent	
Temperature (Atmosphere)	0.07
Pressure (Atmosphere)	0.10
Density (Atmosphere)	2.25
Atmospheric Constituents	10.50
Radar Altimeter	25.00
Electron Density	3.00
	40.92 ~ 41 Watts

	Power
• Surface	
Atmospheric	
Temperature	0.07
Density	2.00
Composition	10.5
Humidity	0.1
Wind Speed and Direction	0.5
Sounds	1.0
Light Level	0.1
Surface Penetrability	0.1
Soil Moisture	25
Seismograph	1.0
Gravity	3.0
Life Detection	7.0
Microscopic Analysis	7.0
Panoramic Mechanism	10.0
TV and Electronics (2 Vidicons, one for Panorama, one for Microscopic Analysis)	20.0

IV. BASE LOAD POWER

		Power
RTG Coolant Pump		15
Orbiter Link	VHF Receiver	2.0
	Command Demodulator	1.8
Earth Link	S-Band Receiver	2.0
	Command Demodulator	1.8
Command Programmer		1.8
Data Handling	Data Processing	3.5
	Buffer Storage	0.5
Power Conversion and Control		10.0
		38.4 ~ 39 Watts

Figure 3.3.5-7. Lander Power Matrix (Continued)



MODE
1
2
3
4
5
400
300
200
100
0
POWER (WATTS)
SEPARATION
REPEAT 12 TIMES PRIOR TO ENTRY
155
155
40
40
40
BEGIN ENTRY PHASE
180
3 MIN
BEGIN DESCENT PHASE
205
IMPACT - BEGIN ORIENTATION
5 MIN
5 MIN
45 MIN
5 MIN
71 MIN
5 MIN
15 MIN
18 HOURS
~10⁶ FT
~30,000 FT.
~9 AM

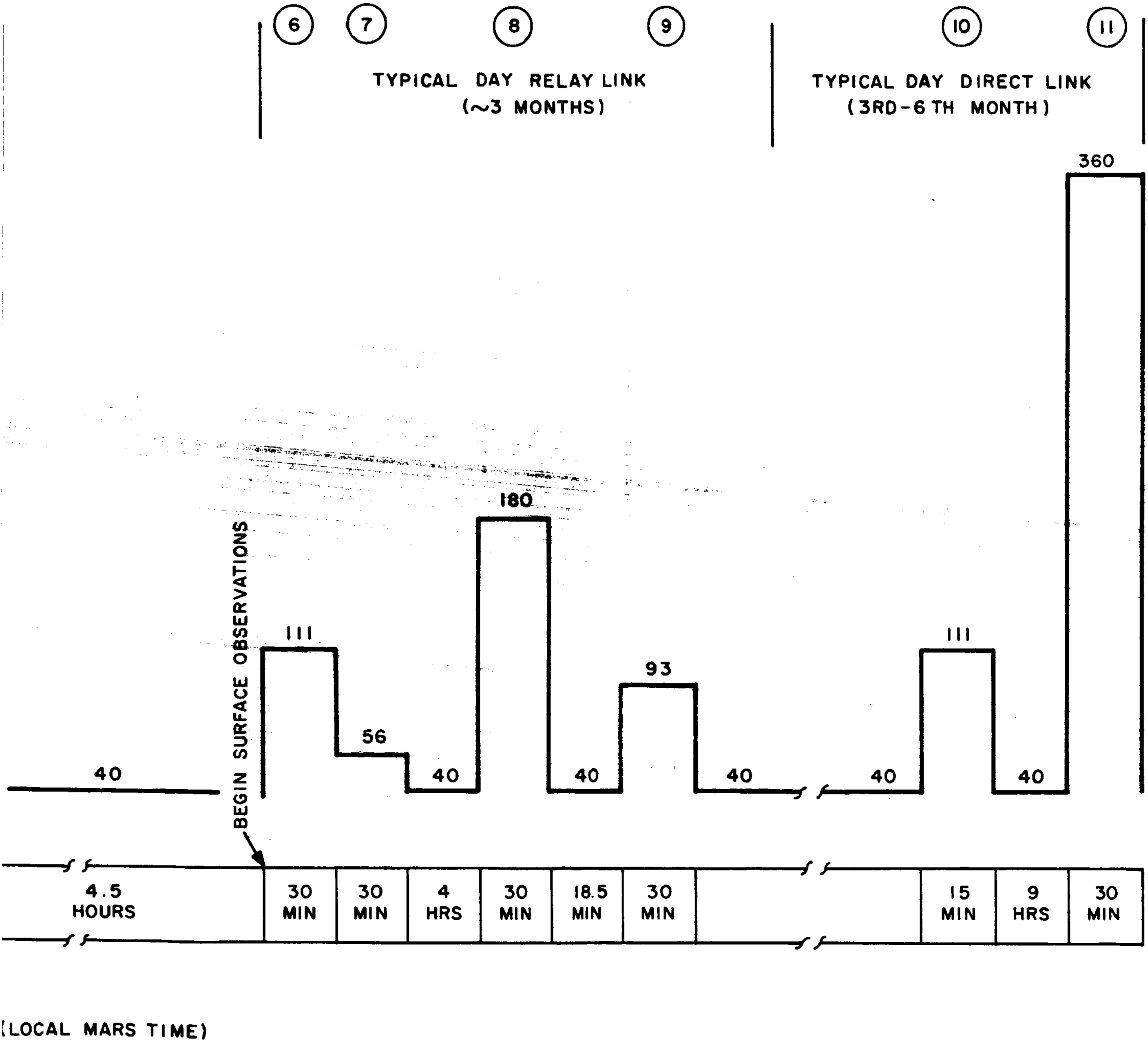


Figure 3.3.5-8. Lander Power Profile

J. Alternate Direct Communication and Power Supply

It is realized that if a continuous low power direct link communication system were operating for approximately ten hours a day, transmitting the same amount of information per day, the power level for the continuous system would be 1/10 of the 70 watts radiated in the present system or 7 watts. A preliminary examination of this concept shows that the data rate per day could be approximately doubled for the same total weight of power supply. In this system, the RTG would be larger and heavier than the one now specified for the Mars Landers and the battery would be lighter. The communication system power is supplied by the RTG alone. Battery charge/discharge losses are eliminated. However, the high gain antenna must either be continually pointed toward Earth or periodically indexed in accordance with beam width requirements. Additional study of this concept is recommended.

3.3.6 DETERMINATION OF MARS 1969 SYSTEM WEIGHTS

With the definition of the power profile and estimate of the communication power requirements, subsystem weights and the system weight schedule could be compiled. Preliminary weight estimates of the system capability had resulted in an estimate of the lander weight. The lander design section then supplied the Orbiter design section with a base diameter and a profile of the lander. Layouts were made of the two landers and the resultant Orbiter structure. The orbiter structure was analyzed for stress, and weight estimates were made. The subsystem weights, including guidance and control, the communication system, and the power supply, were estimated independently. The Orbiter structure also included provisions for the high gain antenna and for the planet horizontal package. Vehicle harness weight was estimated on the basis of total number of components with some allowance for the distance between components. Payload weight estimates were based on the NASA weights for items shown on the NASA payload list supplied by them to MSD. The television system with its optics was a major factor in the total scientific payload weight, but considerable payload was allotted to it because of the important scientific and emotional aspects of the information it could acquire. The propulsion system weight was estimated on the basis of a combustion chamber common to all missions. Pressurization system and fuel tank weights were a function of orbit insertion and midcourse fuel requirements. The orbit insertion fuel factor for the Mars 1969 mission is .904 lbs. of fuel per pound of total orbiting weight. This is based on the chosen orbit of 1 x 19,000 n.mi. and the required velocity increment of 6,400 ft/sec. Midcourse fuel was estimated at 3% of the injected weight. See Table 3.3.6-1 for a listing of the subsystem major components and payload weights.

TABLE 3.3.6-1. SUBSYSTEM WEIGHTS

Sub-System Weights	Mars	1969	Remarks
Structure		418.88	
Orbiter Str.	316.45		
Hardware	40.03		
PHP Str.	56.73		
Hardware	5.67		
Harnessing (Veh.)		106.26	
Power Supply		217.66	
Batteries	21.30		
Electronics	16.10		
Harness (solar array)	7.18		
Fixed Array	173.08		
Guidance & Control		225.85	
Electronics	149.15		
Fe 14 TK & Gas	52.30		
Hardware	24.40		

TABLE 3.3.6-1. SUBSYSTEM WEIGHTS (Continued)

Sub-System Weights	Mars	1969	Remarks
Communications		291.15	
Electronics	259.15		
Antenna (10 ft dish)	32.00		
Diagnostic Instr.		30.00	
Thermal Control		87.00	
Payload		214.62	
* Unidentified	11.82		
Scientific	78.50		
TV	124.30		
Radar Mapper Antenna			
Propulsion		467.41	
Fuel System	364.00		
Pressurization System	103.41		
Orbiting Weight		2058.83	
Orbit Insertion Factor			.904
Orbit Insertion Fuel		1861.77	1 x 19 6400 ft/sec.
Lander Weight		2900.00	2 @ 1450
Midcourse Fuel		210.00	
TOTAL		7030.00	

* Includes Wire Weight

3.3.7 EFFECT OF 10 PERCENT REDUCTION IN SATURN IB PERFORMANCE

NASA requested the G. E. Voyager Study Team to determine the effect on proposed system of a 10% reduction in the Saturn CIB launch vehicle performance. The problem was approached by removing heavy pieces of experimental equipment that made low contributions to the mission value in relation to their weight.

A. Lander

In the Lander, the subsurface drill, pulverizer and sampling handling equipment together weigh 50 lbs. This group of equipment obtains samples of subsurface material and present them for observation to the petrographic microscope and to the life detection group of instruments. Since surface debris can be scraped up and presented to the same instruments by much lighter equipment, (for example, the radioisotope growth detector can acquire its own surface samples with a few ounces of equipment), eliminating these items would only reduce the number and range of available samples of Martian material without either decreasing the number of experiments, or reducing the breadth of determinations to be made in the Martian environment.

The gross weight of each Lander is reduced in proportion to the reduction in payload. For this purpose, Lander payload includes the power supply, instrumentation (including installation and deployment hardware), thermal control and communication subsystems. These items in the Mars 1969 Lander weigh 534 lbs. out of a gross Lander weight of 1450 lbs. Since several Lander sizes were estimated, a curve of Lander gross weight was plotted with respect to Lander payload. This curve is an estimate of total Lander weight on the basis of a vehicle that is designed for a particular payload weight with constant payload density and entry environment for the range of vehicles considered in the study. See the Lander subsystem description in Volume III. The Lander gross weight becomes 1300 lb. when the subsurface sampling equipment is removed.

B. Orbiter

In the Orbiter, the high and medium resolution television and optical systems weigh 95.8 lbs. and contribute pictorial information that only refines and expands the photographic definition of a small portion of the surface area that is already acquired by the low resolution television mapping system. The 20 meter high resolution optics, two 140 meter optics, three image orthicon television cameras with associated electronics were removed from the payload. The revised scientific payload package weighs 143 pounds and provides almost all the mission value obtainable from the original 215 lb. package.

High (20 meter) resolution pictures are lost. Black and white 140 meter pictures can be obtained in the same ratio as for solar and high resolution before, namely, one 140 meter frame per vidicon stereo pair.

The required data rate can be reduced to 60% of the former rate thus decreasing the weight of the hi-gain antenna and the communication subsystem by 4 lbs. The resulting power level, for the power amplifier, of approximately 35 watts leaves the decision to utilize a klystron open to question. The two TPR's are retained.

The power supply solar array area can be reduced approximately 75 watts for a weight reduction of 28 lbs.

The Orbiter structure is reduced by 10 lbs. because of the smaller PHP with the removal of part of the television subsystem. The wiring harness, guidance and control, diagnostic and thermal control subsystems are unchanged.

The weight of the propulsion system hardware for this system is estimated on the basis of the hardware fraction, .184, of the full scale M-69 Mars spacecraft fully fueled propulsion system.

In order to decrease orbit insertion fuel requirements still further, the four end panels of the Orbiter structure are jettisoned prior to orbit insertion. This concept decreases the probability of realizing the specified orbit, because of the jettison event, but if the panels, or a portion of them fail to be jettisoned, the Orbiter can still achieve capture, at the cost of an increase in apoapsis altitude.

These panels weigh 73 lbs. and the net orbiting weight is reduced by this amount. The Orbit insertion fuel factor, .904 lbs. of fuel per pound of orbiting weight is used to estimate orbit insertion fuel with an estimated propulsion system hardware weight. A few iterations and adjustments through the numbers yields the weight schedule shown in Table 3.3.7-1.

TABLE 3.3.7-1. MARS 1969 VOYAGER SYSTEM WEIGHT WITH 10 PERCENT REDUCTION IN SATURN CIB CAPABILITY

Subsystem	Weight
Structure	409
Harness	106
Power Supply	190
Guidance & Control	226
Communication	287
Diagnostic Inst.	30
Thermal Control	87
Payload	143
Propulsion	414
Gross Orbiter	1892
Less Jettisonable Panels	- 73
Net Orbiting Weight	1819
Orbit insertion fuel	1644
Landers	2600
Midcourse fuel	190
Gross Orbiter	1892
TOTAL	6326
Allowable W_i	6327

The three and four digit numbers do not imply that degree of accuracy for this system. The purpose of this analysis is to demonstrate an approach to maintaining as much of the originally specified mission capability as possible with the projected reduction in launch vehicle capability.

3.3.8 MARS 1969 MISSION PROFILE (Figures 3.3.8-1 and 3.3.8-2)

A. Approach

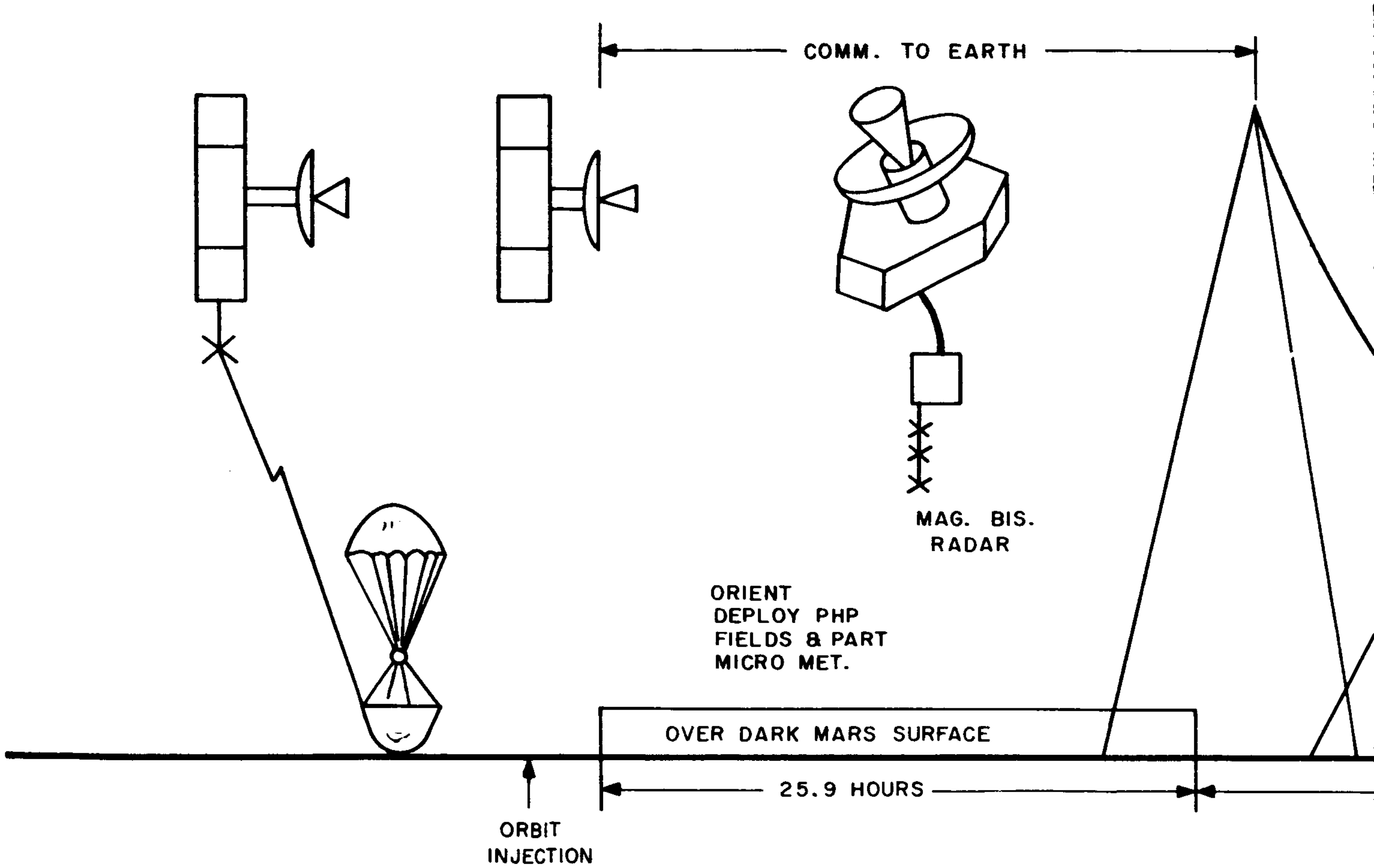
With the successful accomplishment of the transit trajectory with required mid-course corrections, the planet oriented phase of the mission is entered. The terminal guidance observation information is obtained by an image orthicon television camera mounted in the body of the Orbiter so that the planet will be in view with the normal cruise attitude of the spacecraft. This camera is equipped with an appropriate filter and a reticle to provide geometric information based on the photograph of the planet against the star background. This picture is transmitted back to earth through the high gain antenna and the information is used to determine the requirement, if any, for a terminal guidance correction maneuver. If this maneuver is necessary, it is performed approximately 145 hours before encounter to obtain the required orbit perifocus altitude. As the vehicle continues to approach the target planet, additional pictures are transmitted in order to check the accuracy of the terminal guidance maneuver, and to provide information for the computation of the Landers separation maneuvers. Separation maneuvers are planned to take place 17.8 hours before encounter at a distance of 150,000 n.mi. The first Lander to be separated is the one that lands out of the trajectory plane at Syrtis Major.

The second Lander site, Pandorae Fretum, is in the flight plane. If the separation of the first Lander disturbs the vehicle, thus grounding out or over-torqueing the gyros, there is time enough for the vehicle to go back to the sun's orientation and start the whole process over again, based on a re-determination of the gyro settings. The in-plane Lander can be separated as late as 11.8 hours before encounter and still have enough energy in its rocket motor to reach the landing site. During the cruise phase, the Landers periodically telemeter engineering data back to the Orbiter using the VHF relay link on a 71 minute cycle in order to maintain a full charge in the secondary batteries. During separate cruise of the Landers, the Lander power supply and a radioisotope thermoelectric generator are cooled by a radiator which is part of the Lander to Orbiter adapter structure. This adapter is jettisoned just before Lander enters the Martian atmosphere. Cooling during entry is accomplished by a supply of expendable coolant (water) that is boiled off as required.

The line of sight considerations described above require that the Orbiter assume its orbit insertion attitude approximately 40 minutes before injection. In this attitude, a body-mounted antenna provides a necessary beamwidth to enclose the Landers at their landing sites. This is done in orbit insertion attitude to eliminate attitude change maneuver during the short period between the time the Orbiter passes over the Lander site horizon and the time for orbit insertion propulsion burn. Since there is no power available from solar ray during this maneuver, all information relayed from the Landers is stored on the thermoplastic recorder for later transmission.

B. Orbit

After orbit injection burn, the vehicle again assumes sun orientation, performs automatic search for Canopus, and assumes its orbital cruise attitude. The high gain antenna and the planet horizontal package are then deployed as are the bistatic radar antenna and magnetometer boom. Since there is only about 45 minutes



AFTER MAP COMPLETE
BY COMMAND, OVER

LIGHT SURFACE	DARK SURFACE
ADD'L COLOR HI-RES	LONG RANGE LOOK AT POPLAR (M)

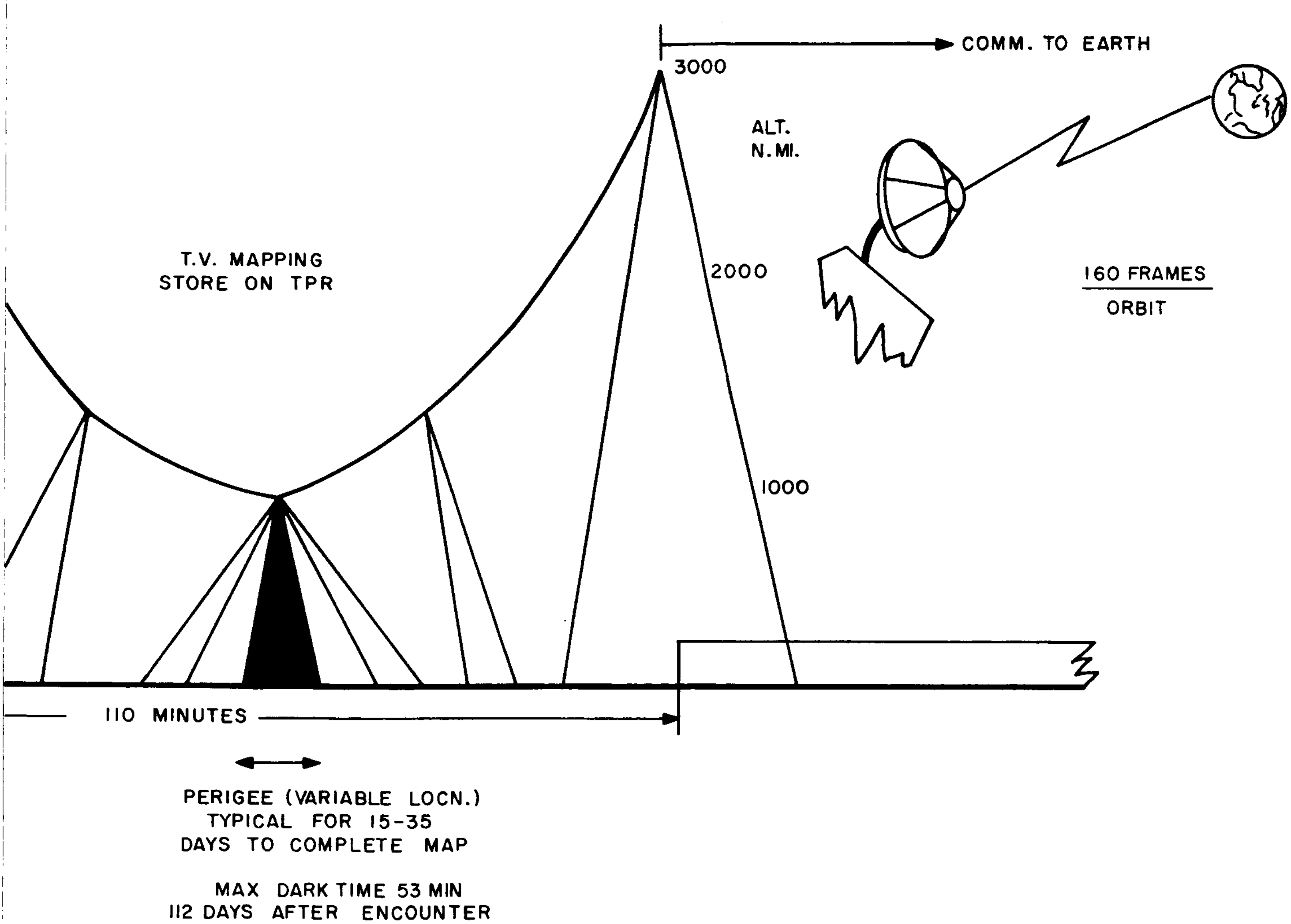
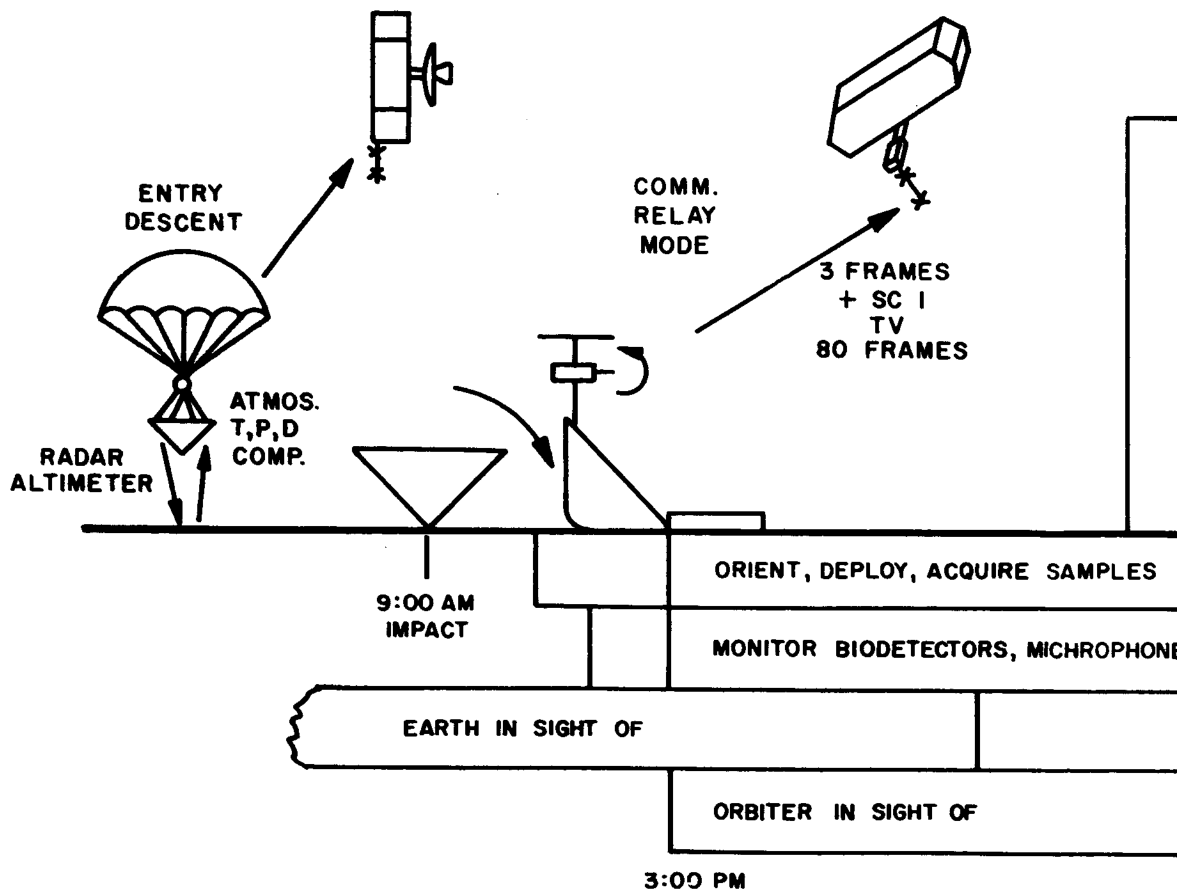
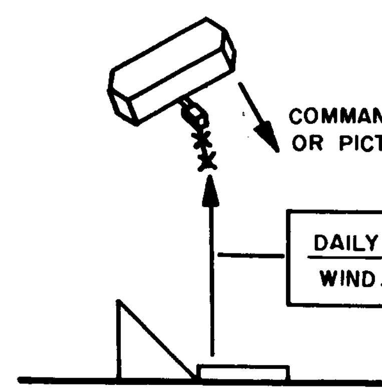


Figure 3.3.8-1. Mars 1969
Orbiter Mission Profile



HOURS AFTER ENCOUNTER	FRAMES TRANSMITTED CUMULATIVE
28	125
58	250
85	363
139	398
168	433
196	508



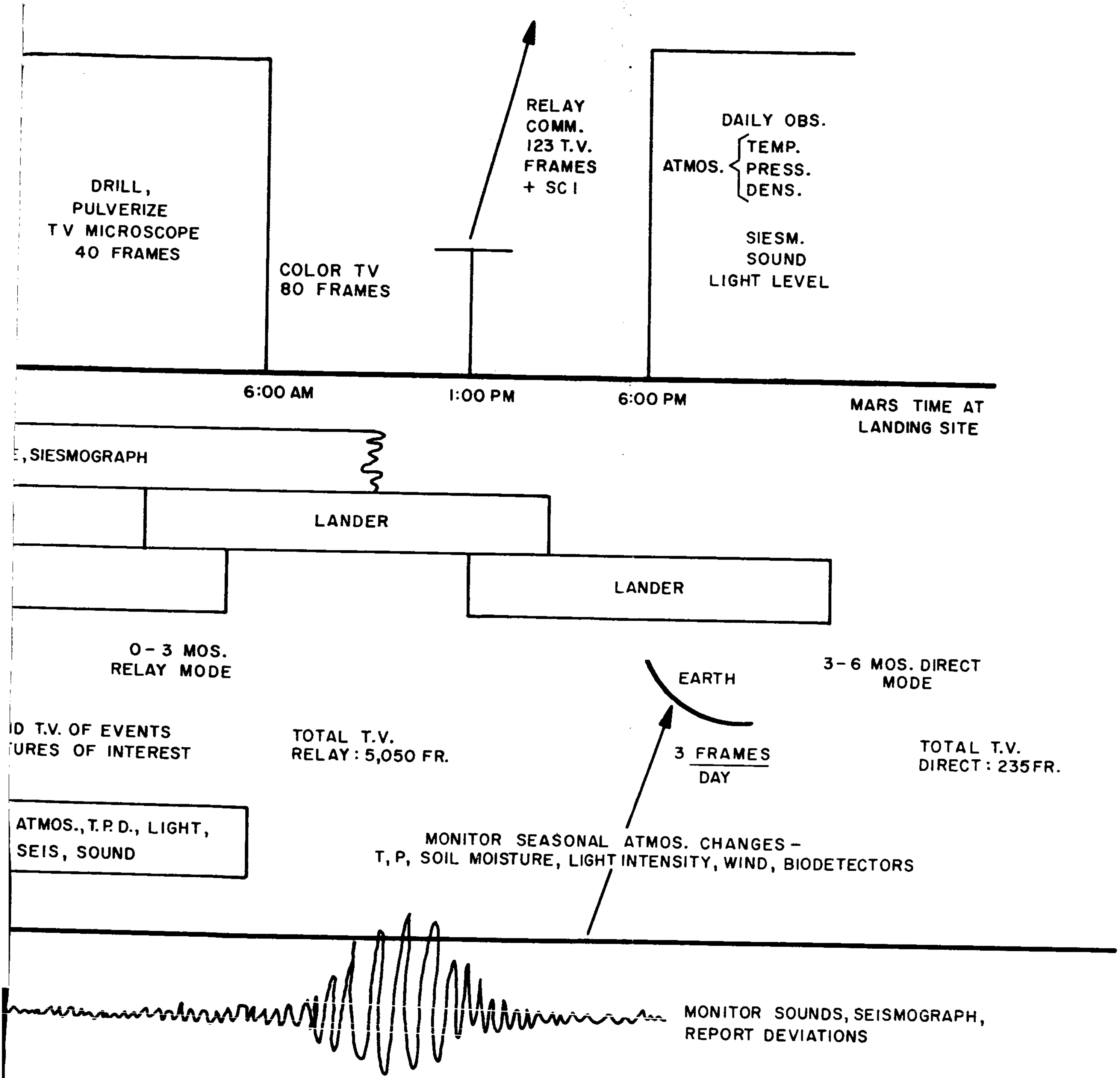


Figure 3.3.8-2. Mars 1969
Lander Mission Profile

from the end of orbit injection burn until the orbiter is over the dark surface of the planet, it is not expected that the deployment sequence will be accomplished soon enough to obtain more than a dozen pictures of the illuminated surface of Mars before the orbiter passes over the terminator. This small amount of television information available in the first pass is insufficient to occupy the capability of the communication system during the subsequent 25.9 hours of the 27.3 hour orbit. Therefore, it is deemed advisable that the image orthicon camera, used to acquire terminal guidance information, be equipped with a mechanism to remove the reticle and filter from the fact of the camera, and that this camera be utilized to take pictures of the planet Mars after Lander separation and before the spacecraft is rotated to the orbit insertion attitude. These pictures will of course be stored in the thermoplastic recorder and could be transmitted after orbit insertion, thereby utilizing otherwise unused communication capacity. There is, in addition, no information available from the Landers beyond the small amount transmitted during their descent phases. If the image orthicon approach guidance camera is used to take pictures of the planet before orbit insertion burn, some pictures of the planet can be acquired even if the orbiter fails to go into orbit. Failure to go into orbit would be sensed by inadequate thrust or no ignition in the engine, and the vehicle would then have an automatic program to deploy the PHP and immediately acquire additional pictures and other information in the fly-by mode. These pictures would be stored on a TPR and, as soon as the spacecraft passed over the terminator, the vehicle would go into cruise attitude thus acquiring power on the solar array, and would transmit the disappointing message and acquired television information to earth.

When the orbiting spacecraft again approaches the terminator, the light sensor in the PHP will initiate the pre-programmed television sequence as described above. During this first orbit both TPR's are not filled with television information. Therefore, additional photographs with the three color image orthicon cameras and high resolution image orthicon cameras will be obtained in order to completely fill the available data storage volume. This is done so that when the Orbiter is transmitting television information during the second and subsequent orbits, hearing conditions can be monitored on earth, and if the error rates are low enough in the received information, the Orbiter communication system can be commanded from earth to increase the rate. Since this increased data rate is more than enough to transmit the regularly scheduled sets of pictures acquired in the mapping pass, the additional pictures stored in the TPR's can be utilized to occupy the capacity of the communication system. The rather large number of television cameras on board the Orbiter increases the probability of obtaining some pictorial information in event of failure of one or more television cameras. If both TPR's are out of commission, an emergency mode of television information to telemetry can be accomplished by utilizing storage ability of the camera tube face and transmitting television information directly from the tube face to earth during the mapping pass. The number of frames obtained per orbit in this back-up mode are greatly reduced and are taken at increased time intervals compared to the standard television mapping sequence.

During the same period, the other instruments are obtaining information from the planet scans and from sensing the fields and particles in the Martian environment. This information is also being recorded during a television pass and being interlaced with the television information during the communication taking place while the Orbiter is over the dark surface of Mars.

C. Lander

Meanwhile, back at the lander sites, temperatures, pressure, density, and composition of the atmosphere are being determined by the sensors on board the landers during their decent. The radar altimeter is recording the altitudes at which these determinations are made in order to improve the definition of the Martian atmosphere. Impact, for the inplane site, is at 9 A. M. local Mars time. After impact, the lander orients itself to the surface and initiates deployment sequences. Sample acquisition mechanisms operate, the VHF antenna and TV camera are deployed, and the first surface information is obtained. The life detection sensors are supplied with surface and atmospheric samples, and the culture chambers are monitored continuously during this portion of the lander operation in order to perceive any sign of life on Mars. The first surface television pictures are also recorded for later transmission. A first opportunity for a relay link occurs when the orbiter appears over the lander horizon at about 3 P. M. local Mars time, and is used even though the range is too far to allow a high data rate. This is done to confirm the operation of the lander. Lander batteries which charge and discharge during descent will have been recharged by this time. During the first night, the subsurface sample is obtained from the first chips from the subsurface drill and is pulverized and presented to the petrographic microscope where survey frames of the samples are taken and recorded.

In the morning, additional surface television pictures are recorded. At 1:00 p. m. the first high data relay period occurs, and 123 television frames and accumulated scientific information are transmitted to the orbiter. Life detectors and multivator chambers have been monitored during this period and this information is also transmitted. Surface sounds and seismograph records have been made thus establishing a base line for these parameters. A monitoring system is initiated so that only deviations from the base lines are transmitted to the orbiter. Determinations are made of atmospheric data such as winds, light levels, temperatures, etc., and they are transmitted as the opportunity arises.

D. Lander-Orbiter Relay Link Operation

Due to the eccentricity of the orbit and the rotation of Mars, relay opportunities vary in time and range. During gaps in relay opportunities, the direct link to earth is actuated in order to confirm its operation and to acquire position information to the lander so it can orient the antenna toward earth. This will occur about the third day on the nominal orbit situation. During the three months life of the orbiter, the relay mode is preferred because of its high data rate enabling the lander to transmit 5,000 television frames to the orbiter during the first 90 days of orbit life. Lander operation during this period includes command and television pictures of areas on particular slides in the microscope, panoramic shots made in varying light conditions, daily atmospheric, and such evidence of seasonal changes as can be obtained by the atmospheric and television sensors. After three months of orbiter life the lander continues to function for an additional three months using a direct mode for information telemetry with a data rate of about 3 frames a day, a total transmission capacity of 235 television frames in this three-month period. The operation of the landers is essentially the same within the limitations caused by the lower data rates.

E. Orbiter Post-Mapping Operation

The orbiter completes its mapping mission in the period of 15 to 35 days after encounter. At this point, if all systems are still functioning adequately, the television system is commanded to repeat the color and high resolution photographs of areas of interest that were detected on the map transmitted during the earlier part of the mission. By commanding the appropriate angles on the PHP, the high resolution camera is directed to attempt pictures of the northern polar regions from along distance out on the eccentric orbit. This additional use of the color cameras and high resolution cameras with their narrow fields of view, increases the acquisition of new pictorial information about the surface of the planet. It is hoped that the high resolution camera will be able to detect the shrinking of the northern polar ice cap as the Orbiter mission proceeds.

F. Mission for Two Spacecraft Launched in Same Opportunity

With the assumption that two complete systems will be assembled and launched from AMR, a certain amount of operational planning is envisioned in the event of the successful insertion into the planetary transit trajectory of both spacecraft. If the first vehicle is successfully injected during the early part of the launch window, it will have accomplished or attempted to accomplish its first midcourse maneuver by the time the second spacecraft is ready to launch. If the first spacecraft successfully executes the first midcourse maneuver, the second spacecraft can be launched on the trajectory which will have as its aiming point the periapsis altitude for the same orbit 1 x 19, 000 n. mi. but injection into the northern hemisphere of Mars instead of the southern hemisphere. If the first spacecraft fails to function at its terminal guidance maneuver, the second spacecraft can be redirected to the southern hemisphere injection because this is of the prime interest as far as lander sites and the Martian seasons are concerned. Thus, if both Orbiters succeed in going into orbit, both hemispheres will be mapped to the 80% of each hemisphere that is possible with the eccentric orbit that we have chosen.

3.4 MARS 1971 SYSTEM

3.4.1 SYSTEM

Since the TV mapping mission on the Mars 1969 orbiter can only obtain a map of 40 percent of the surface of the planet, the 1971 opportunity is essentially a repeat of the mission for the 1969 opportunity. The same orbiter will be inserted in the same 1 x 19 orbit, but with orbit insertion in the northern hemisphere with an orbit inclination of 45°. However, the lower injection energy requirements of the 1971 opportunity and the lower energy requirements for orbit insertion compared to 1969, increase the weight injected into transfer trajectory and reduce the orbit insertion fuel requirements, permitting two 2000 lb. Landers (1450 lbs. in 1969) with increased scientific instrumentation to be carried.

The mission for the 1971 spacecraft is the same as for 1969, except for the choice of hemispheres to be mapped. If the Mars 1969 mission is a complete washout, then in 1971 the entire mission will be repeated as planned for 1969, mapping the southern hemisphere. But if the 1969 spacecraft systems succeed in injecting at least one Orbiter into orbit, then in the Mars 1971 opportunity the spacecraft will be injected into an orbit with periapsis above the northern hemisphere.

Landing sites for the two Landers will of course be chosen on the basis of what was learned in Mars 1969. This does not require any re-design of the Landers themselves. With no mission success at all in 1969, the Srytis Major and Pandora Fretum sites would again be the prime choices. If the 1969 Landers succeed in obtaining surface information and some photographs of the Martian surface without an Orbiter, then the choice of Lander sites in 1971 will be based on earth telescopic information and upon deduction examination of these surface based photographs and on whatever television pictures of the surface of Mars are obtained from the 1969 Orbiter if it performs as a fly-by mission.

If the 1969 Orbiter succeeds in transmitting a complete or partial map of the surface and/or color and high resolution photographs of portions of the surface of Mars, then examination of these photographs may reveal new landing sites of biological or geological interest in the Southern Hemisphere. There will be a 14 month interval between the end of the 1969 Orbiter mission and the launch window for the 1971 mission allowing ample time for data reduction, publishing, examination, and discussion of the photographs. The new landing site selection could even be made after launching the 1971 Voyager spacecraft. However, Lander propulsion requirements may be higher than in 1969 because the spacecraft point of closest approach would lie in the Northern Hemisphere of Mars and the photographic information obtained from the 1969 Orbiter will be in the Southern Hemisphere. If desired, greater Lander propulsion capability would have to be incorporated in the 1971 Landers development program well before the results from the 1969 mission are known.

The 1971 Mars Lander weight estimates in this report do not reflect this possibility of greater propulsion requirements.

3.4.2 SUBSYSTEMS

A. Power Orbiter Supply

The slightly increased power requirements, due to additional Orbiter scientific instruments, resulted in a solar array power level of 446 watts in place of the 440 watts provided in the 1969 Orbiter.

B. Lander Power Supply

The power requirements of the additional scientific equipment carried in this Lander are accommodated by timesharing of the same RTG and battery combination that were employed in the smaller 1969 Lander.

C. Orbiter Communication System

The communication system that was provided in the 1969 Orbiter is used again in 1971. The mean Mars-Earth encounter distance is almost exactly the same in 1971 as it was in 1969. Thus, the expected nominal data rate is the same, (14.25 kilobits per second) as in 1969.

D. Lander Communication System

The 2000 lb 1971 Mars Lander has a larger base diameter than in the 1450 lb 1969 Lander. This additional volume permits incorporation of a helical array antenna with 6.3 db more than in 1969; so that the direct link communication system provides proportionally greater capability than in 1969. The direct link Lander communication system can transmit approximately 1000 Lander television frames in the second 3 months of Lander operation, instead of 235 frames for the 1969 Lander in the same period. The Lander-to-Orbiter relay communication system has the same capability as in 1969.

3.4.3 SYSTEM WEIGHT

Subsystem weights for the Mars 1971 system were modified in accordance with mission requirements, distance from Mars to the Sun for that opportunity, and propulsion requirements. The increased distance from Mars to the Sun requires an increase in power supply weight to compensate for the decrease in performance of the solar cells. The scientific payload is increased slightly from 214.6 lbs. to 223 lbs. These additional weights are easily handled because the orbit insertion fuel requirements are much lower than in 1969 resulting in an orbit insertion fuel factor of 0.475 lbs. of fuel per pound of total orbiting weight for the same 1000 x 19,000 n.mi. orbit. The favorable energy requirements of this opportunity enable the spacecraft to carry two Landers of 2000 lbs each instead of the 1450 lb. Landers of Mars 1969 mission. Midcourse fuel needs are slightly larger because the total injection weight is increased from 7030 lbs. to 7320 lbs. Weight breakdown is shown in Table 3.4.3-1.

3.4.4 MISSION PROFILE

The Mission profile for Mars 1971 is the same as for Mars 1969 with the exception of choice of Lander sites and the mapping of the northern hemispheres. The major difference in operation is due to the increased Lander size which permits the incorporation of much more sophisticated scientific equipment and sample handling equipment.

TABLE 3.4.3-1. WEIGHTS TABULATION

Subsystem Weights	Mars	1971	Remarks
Structure		418.88	
Orbiter Structure	316.45		
Hardware	40.03		
PHP Structure	56.73		
Hardware	5.67		
Harnessing (Vehicle)		106.26	
Power Supply		252.11	
Batteries	21.30		
Electronics	16.25		
Harness	8.23		
(Solar Array)			
Fixed Array	206.33		
Guidance & Control		225.85	
Electronics	149.15		
Fe 14 TK & Gas	52.30		
Hardware	24.40		
Communications		291.15	
Electronics	259.15		
Antenna	32.00		
(10 ft dish)			
Diagnostic Instr.		30.00	
Thermal Control		87.00	
Payload		223.04	
Unidentified*	25.24		
Scientific	73.50		
TV	124.30		
Propulsion		467.41	
Fuel System	364.00		
Pressurization System	103.41		
Orbiting Weight		2101.70	
Orbit Insertion Factor			.475
Orbit Insertion Fuel		998.30	1 x 19 3850 ft/sec.
Lander Weight		4000.00	2 @ 2000
Midcourse Fuel		220.00	
TOTAL		7320.00	

*Includes Wire Weight

3.4.5 ALTERNATE MISSIONS

If a map has been obtained in 1969, if two 1971 Voyager spacecraft are successfully launched, and if one of 1971 Orbiters is successfully injected into the planned orbit with perifocus over the Northern Hemisphere then the second spacecraft could be commanded to execute a terminal guidance maneuver that would change the position of the incoming aerocentric asymptate so that it would intersect the extension of Mars axis of rotation over Mars' North Pole. This would provide a polar orbit with perifocus in daylight over the Northern Hemisphere. It would be desirable to launch the second spacecraft late in the launch window so that the angle, ζ_p, between the asymptote and the planet-sun line will be close to the minimum for the opportunity (see Figure 3.3.5-6). The minimum angle is 73^o which places the polar orbit near the terminator.

This orbit would permit mapping, color, and high resolution photography within the limitations of surface illumination near the terminator of both polar regions. The polar regions cannot be mapped from either the 1969 orbit or the 1971 orbit.

3.5 MARS 1973 SYSTEM

3.5.1 SUMMARY

With the successful completion of two television mapping missions on Mars in the 1969 and 1971 opportunities, the interest in orbital operations should diminish in 1973. The major instrument on board this orbiter is a mass spectrometer to determine the composition of the upper atmosphere of Mars. The orbit is 200 x 9,000 n.mi with an inclination of 53° and the entire spacecraft is to be sterlized.

Since the interest in Lander operations will be growing because of the greater definition of the surface topography and landscape features of the surface of Mars, two 2000 lb. Landers will again be carried in this mission.

Lander sites will be selected on the basis of the most recent information obtained from either the 1969 or the 1971 mission to Mars.

3.5.2 ORBITER

Since the prime purpose of the 1973 Mars Orbiter is the determination of the composition of the upper atmosphere using a mass spectrometer, sample acquisition for this instrument set the periapsis altitude at 200 n.mi thus requiring the entire spacecraft to be sterilized.

Data rate requirements are very low because there is no orbital television equipment in this orbiter. The Klystron is used at the lowest practical power of 35 watts with the 10 ft hi-gain parabolic antenna in order to satisfy the data rate requirement during the terminal guidance observation sequence, when pictures of the planet and star background must be transmitted to Earth. The data rate for this sequence and at encounter is 4.29 kilobits per second. The transmission time for a single terminal guidance image orthicon television frame is 103 seconds, or 17.1 minutes.

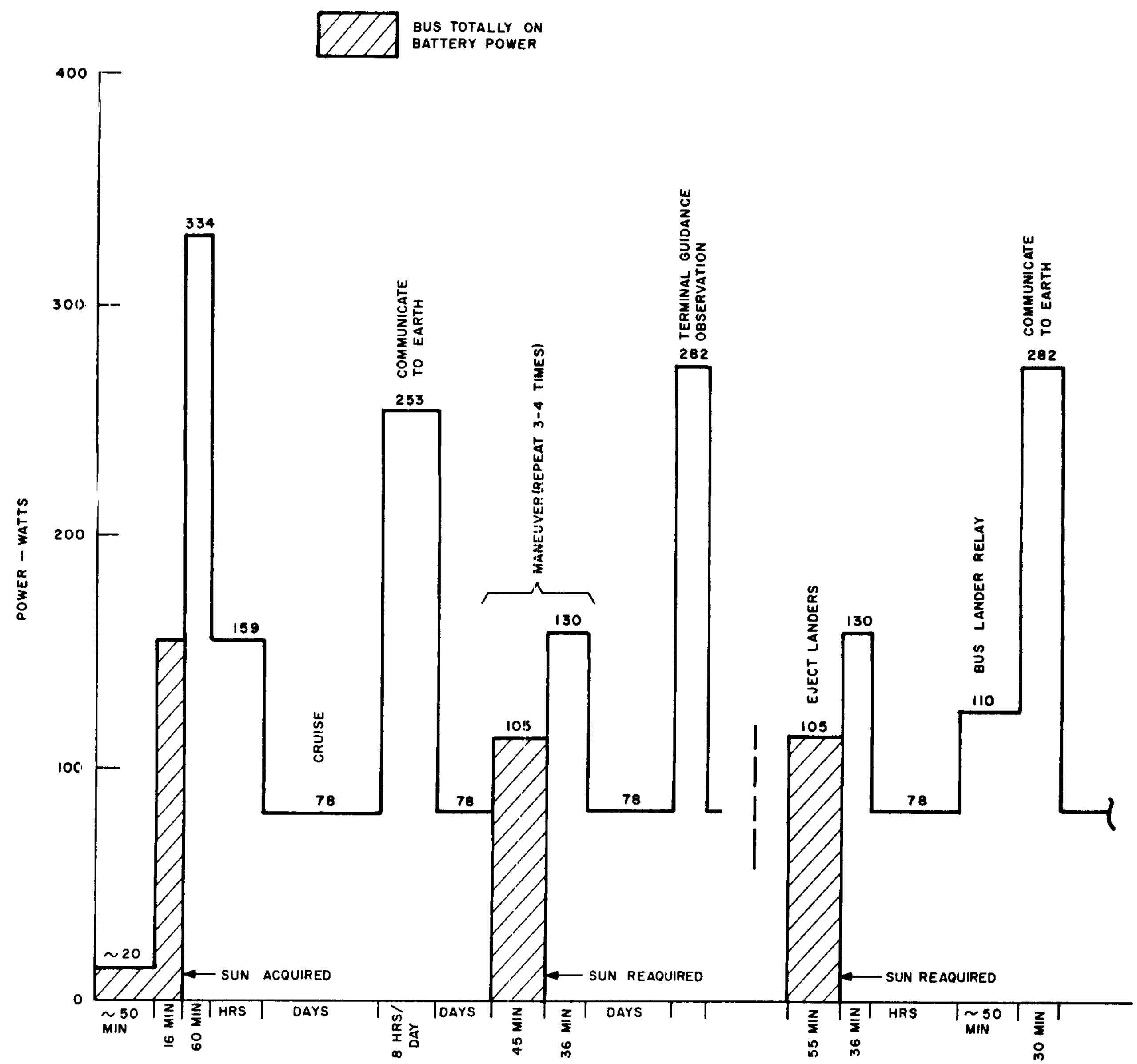
Due to the limited nature of the orbiting mission in 1973, designed orbiting mission life is only ten days and only minimum emergency relay link capabilities are provided in the Orbiter.

The power supply for this spacecraft is a solar array and secondary battery. The array power is 155 watts. The battery weighs 22.7 pounds and provides 122 watt hours of stored energy. Power is supplied to the communication system for intermittent periods of 25 minutes duration by using the secondary battery to supplement the solar array output.

A planet Horizontal Package is not provided for this mission because there is no requirement for planet orientation for any of the instruments.

3.5.3 LANDERS

The 1973 2000 lb. Landers will have a greater probability of entry and impact survival because of the experience with the two prior missions to Mars, and therefore the payloads of the two Landers will be slightly different. The A Lander will carry, in addition to the usual atmospheric sensors, heavier, more complex atmospheric and geological equipment including geophones and explosive charges so that induced as well as natural seismic waves will be obtained in this mission. The Lander will



MARS 1975 BUS POWER PROFILE

also carry a television microscope with less sophisticated sample acquisition equipment than in 1971 and ionospheric profile bottom side sounder and provision to search for meteor ionization trails in the upper atmosphere by means of radar. The B Lander will carry a set of sounding rockets for determination of the characteristics of the upper atmosphere.

The prime method of communication from the Landers to earth will be by direct link with the 70 watt Klystron transmitting through an orientable high gain antenna on board each Lander.

The orientable helical array that was used in the 1971 2000 pound Lander is used again in 1973. Approximately 600 lander TV frames can be transmitted in the first three months of Lander operation.

Lander power supply is the same 70 watt RTG and 29 pound secondary battery used on prior Landers.

3.5.4 SYSTEM WEIGHT

The Orbiter structure is practically the same as for the previous Mars mission; however, the omission of requirements for a planet oriented instrument package eliminate this item and there is a reduction in the Orbiter overall structural weight (refer to Table 3.5.4-1). Harness weight is likewise reduced because of the reduced amount of scientific equipment and the elimination of the PHP.

Low data rate requirements and low data storage requirements reduce the weight of the communication system. The power supply weight is low because of the low data rate and low power requirements of the scientific payload.

The outboard shelves to extend the fixed solar array area are eliminated in this Orbiter. Diagnostic instrumentation weights are reduced because of the smaller number of scientific instruments on board and thermal control system weight is reduced because of the smaller quantity of fuel carried. Scientific payload so far identified amounts to 28 lbs. and there is an allowance for future selection of instruments to repeat individual instruments that failed in previous missions or to accommodate any new instruments available in time for the Mars 1973 spacecraft.

Propulsion system weight is the lowest for the series of spacecraft because of the low weight of the orbiter and reduced orbit insertion fuel requirement. The orbiter insertion fuel factor is only .298 pound fuel per pound of orbiting weight.

3.5.5 MISSION PROFILE

The transit sequence of events is the same as for the Mars 1969 mission with the exception of differences due to 1973 trajectory requirements.

Separate cruise, atmospheric entry and descent phases of this Lander mission will be monitored by the Orbiter using UHF relay link through omnidirectional antennae. The Orbiter will receive UHF diagnostic telemetry from the Lander during entry and descent in the orbit insertion attitude. After orbit insertion the relay link will not be operated except for diagnostic telemetry at very low rates if the helical array antenna on the Lander fails to orient toward Earth.

TABLE 3.5.4-1. SUBSYSTEM WEIGHTS

Subsystem Weights	Mars 1973		Remarks
Structure		338.81	
Orbiter Structure	302.60		
Hardware	36.21		
PHP Structure	0		
Harnessing (Vehicle)		61.07	
Power Supply		106.76	
Batteries	22.70		
Electronics	10.67		
Harness	2.77		
(solar array)			
Fixed Array	70.62		
Guidance and Control		190.35	
Electronics	113.65		
Fe 14 TK and Gas	52.30		
Hardware	24.40		
Communications		217.15	
Electronics	185.15		
Antenna	32.00		
(10 ft. dish)			
Diagnostic Instruments		15.00	
Thermal Control		65.50	
Payload		100.31	
*Unidentified	72.31		
Scientific	28.00		
Propulsion		307.21	
Fuel System	258.80		
Pressurization System	48.41		
Orbiting Weight		1402.16	
Orbit Insertion Factor			.298
Orbit Insertion Fuel		417.84	2 x 9 2550 ft/sec.
Lander Weight		4000.00	2 @ 2000
Midcourse Fuel		180.00	
TOTAL		6000.00	

* Includes Wire Weight

The orbiter is not required to operate after completing the analysis of the upper atmosphere samples obtained at perifocus. Orbiter operation ceases 10 days after insertion.

The two Landers operate independently of the Orbiter. Scientific and sample acquisition equipment is operated with respect to battery charging requirements. Data is transmitted twice a day, with Earth low on one horizon and the low on the other landing site horizon allowing the battery to be recharged between the communication periods.

The "B" Lander is programmed to send up its sounding rockets at different times in the day and night in order to detect diurnal changes in the upper atmosphere.

3.5.6 MARS 1973 ALTERNATE SYSTEMS NO. 1

The failure to acquire a television map from an orbiting spacecraft in either Mars 1969 or the Mars 1971 missions would cause the substitution of an alternate mission for the Mars 1973 low periapsis altitude atmospheric sampling mission described above. The spacecraft designed for the 1971 opportunity is used for this mission with slight modifications to the solar array based upon the Mars sun distance for this opportunity. The somewhat increased orbit insertion fuel requirements for the same television mapping of 1 x 19,000 n.mi. would reduce the available Lander weight to 2,900 lbs. and therefore two 1450 lb. Landers as designed for the 1969 opportunity would be carried on this mission.

Mission profile for this alternate spacecraft for the 1973 opportunity is precisely the same as for the Mars 1969 and 1971 television mapping missions. Table 3.5.6-1 lists the subsystem weights and orbit insertion parameters.

3.5.7 MARS 1973 ALTERNATE SYSTEMS NO. 2

Future manned landings on Mars will require pictures of possible landing sites with a minimum optical resolution of one (1) meter. The size and weight of television and optical systems to provide this resolution from the periapsis altitude of 1000 n.mi. of prior missions is prohibitive. However, the low periapsis altitude, 200 n.mi., of the orbit selected for the 1973 Mars mission could be a useful approach to obtaining the one meter resolution.

An analysis of the television and optical parameters showed that, for example, one meter resolution could be obtained from an altitude of 120 n.mi. with a television camera and optical systems that weighs 117 pounds.

If a periapsis altitude of 120 n.mi. is too low for direct insertion because of the effects of guidance accuracy, then the sterilized Orbiter must be inserted in a highly eccentric orbit with the typical periapsis altitude of 1000 n.mi. It is proposed that this Orbiter be equipped with a liquid, storable propellent propulsion system with a capability of at least two starts. After the initial, direct insertion, orbit has been accurately defined, the auxiliary low thrust, propulsion system would be fired once, or if required, twice, at apoapsis in order to accurately reduce the periapsis altitude to the required 120 n.mi. The restart capability permits a two-step velocity increment. The required velocity increment is estimated to be 260 feet per second to reduce the periapsis altitude of a 1000 x 19,000 n.mi. orbit to 120 n.mi. The weight of this propulsion system, including thrust chamber, tanks, valves, fuel and operating controls is estimated to be 119 lbs. The combined weight of this propulsion and

TABLE 3.5.6-1. SUBSYSTEM WEIGHTS AND ORBIT INSERTION PARAMETERS

Subsystem Weights	Mars 1973 (Alt)		Remarks
Structure		418.88	
Orbiter Structure	316.45		
Hardware	40.03		
PHP Structure	56.73		
Hardware	5.67		
Harnessing (Vehicle)		106.26	
Power Supply		235.88	
Batteries	21.30		
Electronics	16.25		
Harness (solar array)	7.72		
Fixed Array	190.61		
Guidance and Control		225.85	
Electronics	149.15		
Fe 14 TK and Gas	52.30		
Hardware	24.40		
Communications		291.15	
Electronics	259.15		
Antenna (10 ft. dish)	32.00		
Diagnostic Instruments		30.00	
Thermal Control		87.00	
Payload		235.41	
*Unidentified	37.61		
Scientific	73.50		
TV	124.30		
Propulsion		360.99	
Fuel System	294.28		
Pressurization System	66.71		
Orbiting Weight		1991.42	
Orbit Insertion Factor			.515
Orbit Insertion Fuel		1025.58	1 x 19 4150 ft/sec.
Lander Weight		2900.00	2 @ 1450
Midcourse Fuel		183.00	
TOTAL		6100.00	

* Includes Wire Weight

TV/optical system is well within the payload capability of the 1973 Mars spacecraft that would be allowed by the initial 1000 x 19000 n.mi orbit insertion fuel requirements.

This concept is suggested as Mars 1973 alternate No. 2.

A study of the effects of realistic guidance accuracy on these low periapsis altitude missions should be made in order to determine the necessity for an auxiliary vernier orbit trim propulsion system.

3.6 1975 MARS SYSTEM

3.6.1 SUMMARY

With the successful completion of three orbiting missions, two television mapping and one atmospheric sampling mission, the interest in orbiting operations should have diminished by 1975. Therefore, this high energy opportunity will not include an orbiting mission, but the orbiter will be utilized as a flyby bus to deliver two 2000 lb. Landers to the surface of Mars. The mission of these Landers will be to obtain a more detailed determination of the characteristics of the Mars surface and subsurface environment utilizing the payload capacity of these large Landers to deliver heavy, sophisticated lasers or other advanced payloads which are just now beginning to be defined. An alternate payload for at least one of the two 2000 lb. Mars 1975 Landers is a surface roving vehicle which is capable of exploring in detail a larger area on Mars surface than can be viewed by a stationary lander.

3.6.2 SYSTEM WEIGHT

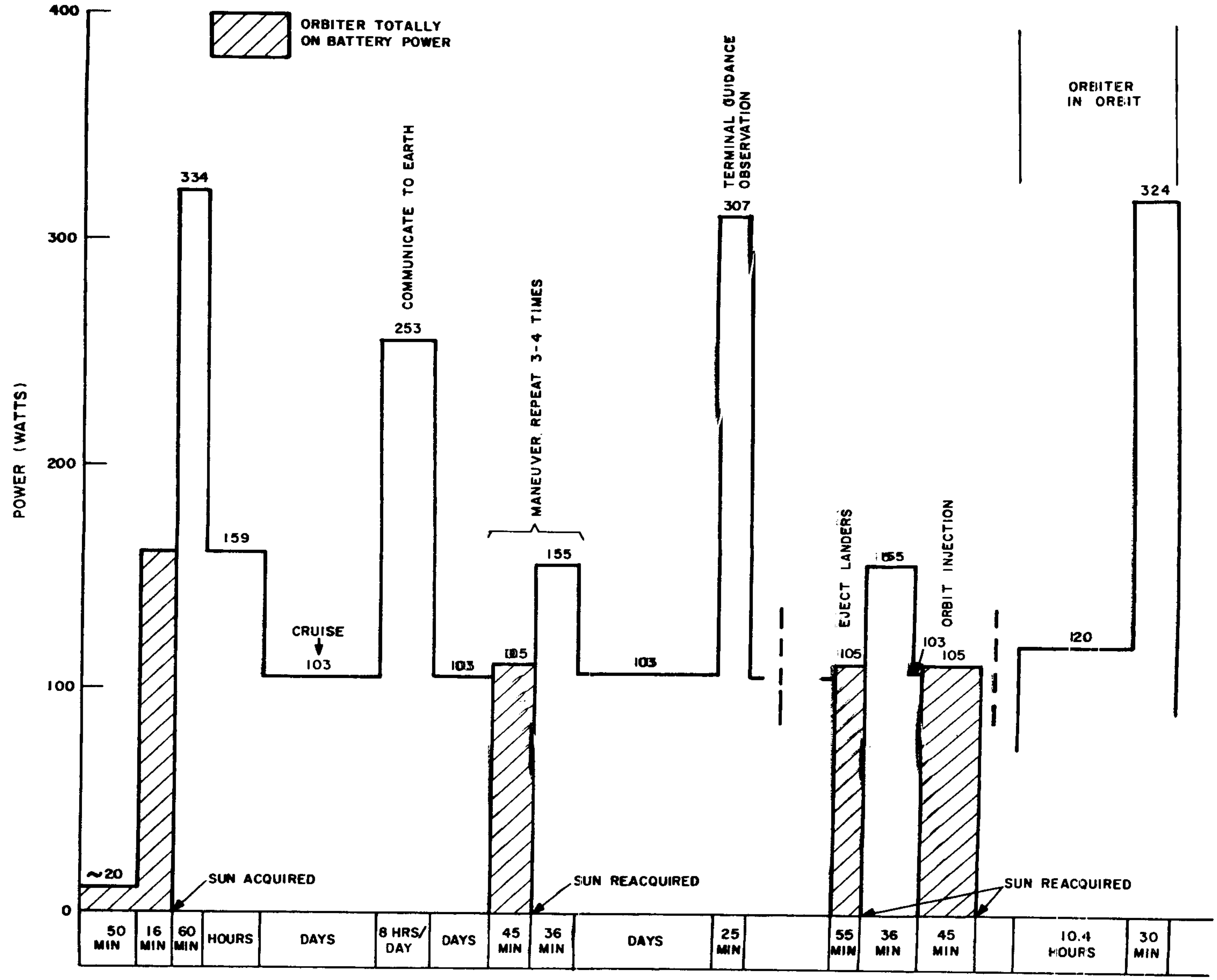
The weights for the Mars 1975 System are given in Table 3.6.2-1. The flyby bus structure is substantially the same as prior Orbiter except the planet oriented instrument package will not be needed. The vehicle harness weights are reduced to the lowest weight (54.7 lbs.) for this series of spacecraft, because of the complete absence of scientific equipment on board the flyby bus. Communication requirements are quite minimal. The only significant requirement is the ability to transmit in reasonable length of time the image orthicon television pictures used during terminal guidance observation sequence. Peak power from the solar array is 130 watts, and total power supply weight is also a low of 90 lbs. In this mission, the secondary batteries on board each Lander are used to supply power for the spacecraft during maneuvers and during communication peak loads. Communication to earth will be by means of a 35 watt klystron through the 10 ft. parabolic antenna. All required solar cells are mounted on the bottom panel of the orbiter structure and the perimeter extensions are not required for this mission. The guidance and control subsystem weight is reduced by the elimination of electronics and actuation hardware from the Planet Horizontal Package. Diagnostic instrumentation is a minimum of 15 lbs. because of the small number of subsystems and components that must be monitored. Thermal control is reduced to 65.5 lbs. because of the small amount of fuel carried and the low number of scientific instruments. There is provision for 60 lbs. of payload which may be utilized for any transit or planetary environmental scientific observations that might be desired at that time. The propulsion system weight is the same as for Mars 1973, and is obviously over-designed for the mission which requires only the performance of midcourse and terminal guidance correction maneuvers. This is a weight penalty brought about by common use of subsystems from mission to mission. If the mission evolves in such a way as to require optimization of some of these common subsystems, then it can easily be accomplished before the 1975 opportunity.

3.6.3 MISSION PROFILE

In order to eliminate the need to sterilize this flyby bus, the entire spacecraft with the Landers attached will not be placed on an impact trajectory. Therefore, the Landers must have propulsion capability in order to change the flight path from a

TABLE 3.6.2-1. SUBSYSTEM WEIGHTS

Subsystem Weights	Mars	1975
Structure		337.44
Orbiter Structure	302.60	
Hardware	34.84	
PHP Structure Hardware	0	
Harnessing (Vehicle)		54.70
Power Supply		87.09
Batteries	8.60	
Electronics	10.26	
Harness (solar array)	2.30	
Fixed Array	65.93	
Guidance & Control		190.35
Electronics	113.65	
Fe 14 TK & Gas	52.30	
Hardware	24.40	
Communications		217.15
Electronics	185.15	
Antenna (10 ft dish)	32.00	
Diagnostic Instr.		15.00
Thermal Control		65.50
Payload		60.56
Scientific	60.56	
Propulsion		307.21
Fuel System	258.80	
Pressurization System	48.41	
Orbiting Weight		1335.00 (flyby)
Orbit Insertion Fuel		
Lander Weight		4000.00 2 @ 2000
Midcourse Fuel		165.00
TOTAL		5500.00 lbs.



MARS 1973 ORBITER POWER PROFILE

flyby to a trajectory impact after Landers are separated from the bus. Therefore, the sequence of events for this mission is exactly the same as for the basic 1969 Mars system up to orbit insertion. After the Landers have separated from the flybybus, diagnostic telemetry is transmitted from the cruising Landers to the bus. An omni-directional VHF antenna is mounted on the bus to receive the VHF transmission from the Landers as they enter the atmosphere. The bus will be in the cruise attitude and can relay the diagnostic information to earth.

After the Landers impact, the programming and control units command the deployment of the high gain earth antenna, the panorama television subsystem, the wind speed and direction sensor, and the various scientific instruments.

The power supply and communications subsystems of the Mars 1975 Lander are the same as those utilized on the Landers in the 1971 and 1973 Mars missions. After impact, the only telemetry link is the 70 watt klystron transmitting through the orientable helical array direct to earth.

3.6.4 ALTERNATE PAYLOAD

A surface roving vehicle is proposed as an alternate payload for at least one of the 1975 Landers. Rovers can provide almost unlimited sampling sites for petrographic, biological and geological instruments. A roving vehicle can transport a television camera to a preselected area and by moving the camera over the area, can provide the one-meter resolution required to define a future manned landing site.

By this time, future plans for manned landings on Mars may be maturing enough to require landing site definition, and command and control techniques may have progressed far enough to assure reliable operation of a sophisticated and complicated device at interplanetary ranges.

Incorporating a roving vehicle in one of the 1975 Landers would require the redesign of the interior structure. Other payloads would probably be eliminated.

The guidance would be by programmed commands based on televised guidance information. Surface speed would be measured in feet per hour because of the communication delay. Power supply would probably be a radioisotope thermoelectric generator. Communication could be direct or by relay through the Lander communication system.

Two approaches to the design of this roving vehicle are suggested: a) a small one, with only a relay communication link to the main Lander, and thus limited to operating range within line of sight of the Lander antenna, and b) a larger one which occupies a greater portion of the interior volume of the Lander with all the power supply, communication equipment and scientific payload on board the roving vehicle. When this vehicle leaves the Lander, only an empty shell remains at the landing site.

The large Lander would not be range limited, but would have to carry a heavy communications system in order to have a direct link from the roving vehicle to the earth. Most of the scientific payload in the Lander would move with the vehicle in order to reach new sampling sites. Additional study of this concept is suggested.

3.7 VENUS 1967 SYSTEM

Initially the emphasis in the study was placed on the design of a system for the Venus 1967 opportunity. However, it quickly became evident that there will be insufficient time to develop a reliable Voyager spacecraft for the June 1967 launch window. Therefore the emphasis was shifted to the system design for the Mars 1969 opportunity.

As a matter of record the concepts that were considered are given in Section 2.6, with the system employing two small 500 lb. Landers and a radar orbiter being the most attractive for the first Venus mission. A system employing a large Lander must await a more exact definition of the Venusian atmosphere and surface properties.

3.8 VENUS 1970 SYSTEM

3.8.1 SUMMARY

The system aspects of the 1970 Voyager mission to the planet Venus are dominated by the Venusian atmosphere. The inclusion of a reasonably high resolution radar mapping system in the orbiter scientific payload is indicated by the continuous cloud cover. The extremely high Venusian surface temperatures restrict the operating time of the Venusian lander to the limit of the capability of a stored quantity of coolant in the lander. A rather high orbit insertion fuel factor, even for eccentric orbits, along with the heavy thermal control systems in landers preclude two landers on a single spacecraft for the 1970 Venus mission. Communication and power requirements are set by the radar information and by the need to have sufficient data rate at the end of the radar mapping mission to handle the quantity of the data acquired per orbit at that time. The surface survival time of the single minimum 1970 Venus lander is nominally ten minutes. Along with the radar, television equipment is provided in the orbiter and in the lander to obtain pictorial information about the nature of the Venusian cloud layers. The Venus lander is equipped with descent radar to aid in determination of the elevation of these cloud layers.

3.8.2 RADAR SYSTEM

Since the side looking synthetic aperture radar mapping system would provide quantities of information that would design the communication and power supply systems, a parametric study of radar system weights was initiated. The results indicated that a nominal 2 x 2 n.mi. resolution would be the practical limit for a 1000 x 2,250 n.mi. orbit. A more detailed study of the requirements of the radar is given in Section 3.0 of Volume III.

A more eccentric orbit was chosen so the spacecraft could carry at least one lander. This orbit has a 1000 n.mi. periapsis and a 4,300 n.mi. apoapsis with a period of 3.57 hours. Since the seasonal progression of the planet Venus could cause 15.1 n.mi. of the surface of Venus to pass underneath the orbiter for each orbit, the swath-width of the radar system, 112 n.mi., would require that a complete radar map be obtained in onlyone out of seven passes. In order to reduce the size of the battery so that the radar determination can be made while the orbiter is passing through Venus in shadow, the requirement for radar coverage of one out of seven passes is distributed so that the radar equipment is powered 1/7 of each pass. The maximum operational altitude of the radar system is 2,500 n.mi. Since the orbiter is below 2,500 n.mi. for approximately 50 minutes per orbit, its power is required for the radar subsystem for 1/7 of 50 minutes or approximately 7 minutes per orbit.

Due to the approach geometry, the periapsis of this orbit will be at a latitude of about 40° below the "equatorial" plane of Venus. Since the rotational characteristics of Venus are in doubt, the equatorial plane is defined as the intersection of the Venus orbital plane with the planet Venus. Seasonal progression of the orbit around Venus will be regarded as being rotation about an axis perpendicular to the Venus orbit plane through the center of the planet. Due to the position and eccentricity of this orbit, the orbiter will pass over the south pole within the 2500 n.mi. operating limit of the radar system but will be out of the range of the north pole.

In order to increase the coverage of the radar mapping mission, the radar is directed to one side of the orbit and then to the other on subsequent passes, so that the area is mapped by the side looking radar ahead of the progression and behind the progression of the orbit around the planet Venus. This increases the available radar mapping area and requires that the radar be supplied with power for 14 minutes per orbit instead of 7. It also doubles the data rate during the first 25 days. Since this is done only at the beginning of the mission when the communication range is the shortest, the available data rate is sufficient.

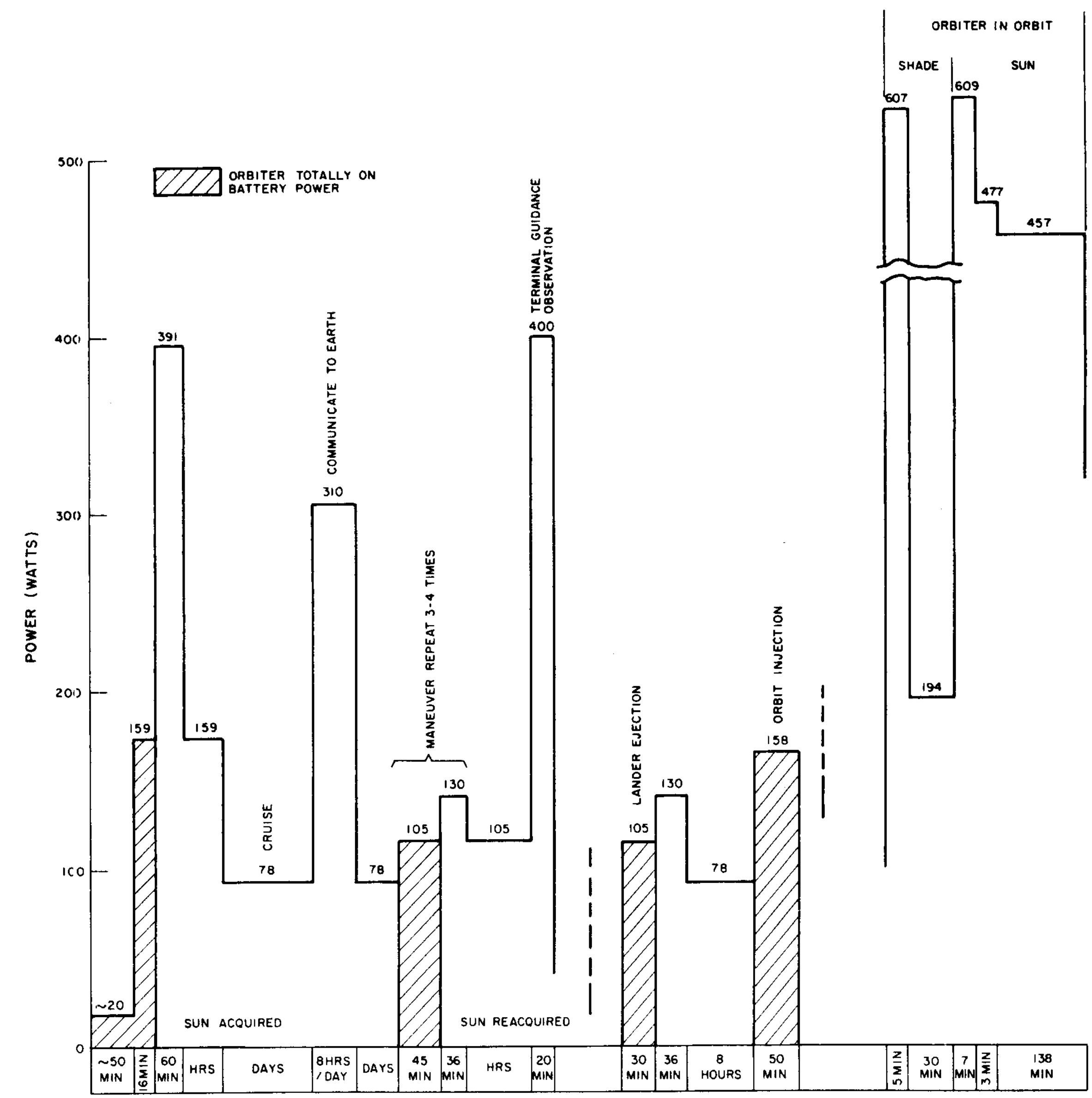
3.8.3 POWER

The requirement for this mode can be met because the Orbiter does not enter any shadow during the first 25 days of the mission. The radar power requirement of 440 watts plus the attitude control or the housekeeping needs of the orbiter establish a peak power requirement of 609 watts. The battery is sized to provide this power for six minutes during a shadow time of approximately 35 minutes and to supply the housekeeping power during the remainder of the shadow time. This battery weight is 47 lbs. Communication with the earth takes place only when the Orbiter has sunlight available for power, and the earth is also not occluded by the planet Venus. Separate high gain parabolic antennae are provided for both the radar mapping system and for the communication system. A small amount of scientific equipment is carried in the PHP which is mounted behind the reflector of the radar antenna so that the instruments will view the planet and be aligned with the local vertical when the antenna is aligned to the required side looking angle for the radar system.

A minimum television system including the incorporation of a color wheel and small optics system with a vidicon camera is also provided in the planet oriented portion of the radar antenna assembly. It is noted here that the image orthicon camera utilized for approach guidance and observation would be carried into orbit in this mission. Therefore, there is an incentive to design the installation of the image orthicon camera so that it could also be used in orbit.

The communication system of this orbiter is also equipped with VHF relay link equipment to acquire data transmitted from the lander. Since the lander's survival time is so short, information can be acquired from the lander only while the orbiter is in the orbit insertion attitude. Since the radar antenna and Planet Horizontal Package will not be oriented to the local vertical at this time, the VHF high gain yagi antenna, which was used in the Mars 1969 system, is mounted on the body of the orbiter with a single axis actuator so it can be continually oriented toward the Lander. This arrangement provides relay link data rate as high as 50,000 bits per second or 6 TV frames per minute. Due to the short surface survival time of the Venus lander, no VHF receiving equipment will be incorporated in this Lander and no attempt will be made to modify surface operations during the lander mission by command from the orbiter.

The Lander is equipped with a vidicon television camera which is able to see through the aft end of the Lander after the cover is jettisoned. An angular view past the aft cover acting as a drag body can be taken of the under side of the Venusian cloud cover. When the Lander has been stabilized on the surface of Venus a moveable prism provides pictures of the surface near the lander through a limited field of view. All power requirements of the Lander are fulfilled by primary batteries which is the lightest method for this one short duration mission.



VENUS 1970 ORBITER POWER PROFILE

3.8.4 MISSION PROFILE

The transit portion of the Venus 1970 mission utilizes the same events and equipment that is provided for the basic Mars 1969 system, although modified for distances and times required for the planetary transfer to Venus. Because only one lander is carried, the requirement for separation of a second Lander is eliminated. In order to achieve line of sight time the lander will have to be separated at a much greater distance and with 700 ft. per sec. tangential velocity compared to 400 ft. per sec. for the Mars landers. This is caused by the higher hyperbolic excess velocity of the spacecraft with reference to Venus as compared to the excess velocity in the Mars mission. The relay link is operated only during the orbit insertion attitude period as described above. It is recognized that the probability of the orbiter operation surviving without component failure for the required 200 days is fairly low and the radar mapping system can chart only about 75% of the surface in any case. Therefore, an additional radar mission would have to be flown in future opportunities if a map of the remainder of the planet is desired. This can be modified if both 1970 spacecrafts succeed in achieving an orbit. Since the rotational period of Venus is open to question, and in any case moves very slowly, orbit trim propulsion capability is not provided in this orbiter. If the radar returns reveal interesting anomolies in the surface of Venus any additional looks at these areas of interest must be performed very shortly after the information is received because of the constant progression of the orbit around the planet. In this case, the side looking synthetic aperture radar is looking in the direction of orbit progression, and therefore an opportunity to take a second look at areas of interest will exist some 25 days later when the radar antenna can be oriented to the other side of the orbiting spacecraft and directed by earth command to perform this second look is desired.

3.8.5 SYSTEM WEIGHT

The weight of the orbital structure for the 1970 Venus spacecraft (Table 3.8.5-1) shows a slight weight penalty due to the use of the structure designed for Mars Orbiters which carry two Landers. The top of the orbiter structure is reinforced to handle the more concentrated load of the single lander mode in the center. However, this is compensated for in part by the reduced weight of the Planet Horizontal Package structure, which is a part of the parabolic radar antenna. Weight of the power supply is based upon the requirements shown in the power profile. The orbiter structure is large enough so that all the required solar cells can be mounted within the outer limits of the structure, thus eliminating the fixed array extensions used in the 1969 and 1971 Mars systems.

The communications system uses the same 50 watt klystron that was used in the Mars 1969 and 1971 systems. A dish size of 10 feet was not optimum for this mission because of the somewhat lower specific weight of the power supply, but a slight penalty is accepted in order to provide for use of a system already developed and tested.

The guidance and control subsystem components are the same as for the Mars missions. However, the total impulse provided for the attitude control is somewhat higher than for Mars because of higher gravity losses in orbit.

TABLE 3.8.5-1. SUBSYSTEM WEIGHTS

Subsystem Weights	Venus 1970	
Structure		414.55
Orbiter Structure	339.37	
Hardware	41.74	
PHP Structure	30.40	
Hardware	3.09	
Harnessing (Vehicle)		76.95
Power Supply		202.59
Batteries	46.80	
Electronics	17.55	
Harness (solar array)	5.67	
Fixed Array	132.57	
Guidance & Control		231.05
Electronics	149.15	
Fe 14 TK & Gas	57.50	
Hardware	24.40	
Communications		226.65
Electronics	194.65	
Antenna (10' dish)	32.00	
Diagnostic Instr.		30.00
Thermal Control		87.00
Payload		137.00
Scientific	37.50	
TV	10.50	
Radar Mapper	57.00	
Antenna	32.00	
Propulsion		739.21
Fuel System	541.80	
Pressurization System	197.41	
Orbiting Weight		2145.00
Orbit Insertion Fuel		4372.00
Lander Weight		525.00
Midcourse Fuel		218.00
TOTAL		7260.00

The propulsion system is based on the orbital insertion requirements. It is the heaviest system of the series, 740 lbs., because of the large quantity of fuel, 4172 lbs., carried in this series of spacecraft. The single lander weighs a total of 525 lbs. and is regarded as the minimum Lander that should be consided.

NOTE: While this system shows an orbiting life of 90 days, it is recognized that this would not be sufficient for the radar mapper to obtain a complete map of the planet. If reliability studies of the extended duration orbiting mission show that it is worthwhile to extend mission life, the supply of attitude control gas would have to be increased to approximately 80 lbs. and the orbit eccentricity would have to be increased to accommodate this increase in weight and decrease in available orbit insertion fuel.

3.9 VENUS 1972 SYSTEM

3.9.1 SUMMARY

Combination of a Type II trajectory with injected weight of 7250 lbs. and a rather eccentric orbit and a favorable orbit insertion fuel requirement for this opportunity presents the possibility of carrying a larger Lander to the surface of Venus. Only a radar altimeter is carried on this Orbiter thus reducing the weight and power requirements from the 1970 Voyager system. Additional instruments to obtain information on the properties and again a small vidicon television system are incorporated in the Venus 1972 Orbiter. The landing weight is now sufficient to allow the Lander to survive for one complete orbit period plus the time the Orbiter requires to pass from periapsis to the lander site horizon.

3.9.2 MISSION PROFILE

Mission profile for this system is the same as for the 1970 Venus opportunity with the exception of the effect of the survival time of the Landers. After Lander separation is completed the Orbiter returns to the cruise attitude to obtain solar power to recharge its secondary batteries and communicate the results of the separation sequence to Earth. The image orthicon camera used for terminal guidance observation is again utilized to obtain approach pictures of the planet. This television information is stored in the TPR recorder because the high-gain antenna is now stowed in preparation for the acceleration forces to be experienced during orbit insertion. Approximately 25 minutes before orbit insertion the vehicle assumes the orbit insertion attitude in order to orient an antenna mounted on the body of the Orbiter so that its beamwidth will cover the Lander sites. This antenna is a yagi high-gain VHF antenna and is provided with a single degree of freedom actuator that will continually orient the antenna to the coverage of the Lander site as the Orbiter proceeds towards periapsis. At this time the PHP is not deployed and even if a yagi antenna were mounted on the PHP as it is in the Mars Orbiter, it would not be available during the orbit insertion pass over the Lander site. Slewing the body-mounted yagi antenna is required because at the low periapsis altitude the beam-width of this antenna is too narrow to include the entire surface of the planet. After orbit insertion the PHP and high-gain antenna for the earth link are deployed and the Orbiter proceeds on its scientific mission. Towards the end of the first orbit the Orbiter again assumes the orbit insertion attitude so that the body-mounted yagi antenna can again be oriented towards the lander site. Due to the expected uncertainty in orbit apoapsis altitude, a signal from the television photometer indicating the terminator on the way towards the lander site will initiate the yagi antenna slewing sequence. In order to obtain maximum use of available line of sight time between Orbiter and Lander, a VHF signal will be transmitted from the Orbiter towards the Lander continuously and as soon as the Orbiter passes over the Lander site horizon the VHF receiver on board the Lander will acquire this signal and command the transmitter onboard the Lander to commence operation. The data rate of this transmission can be altered in order to take a maximum advantage of the variable range during the line of sight opportunity.

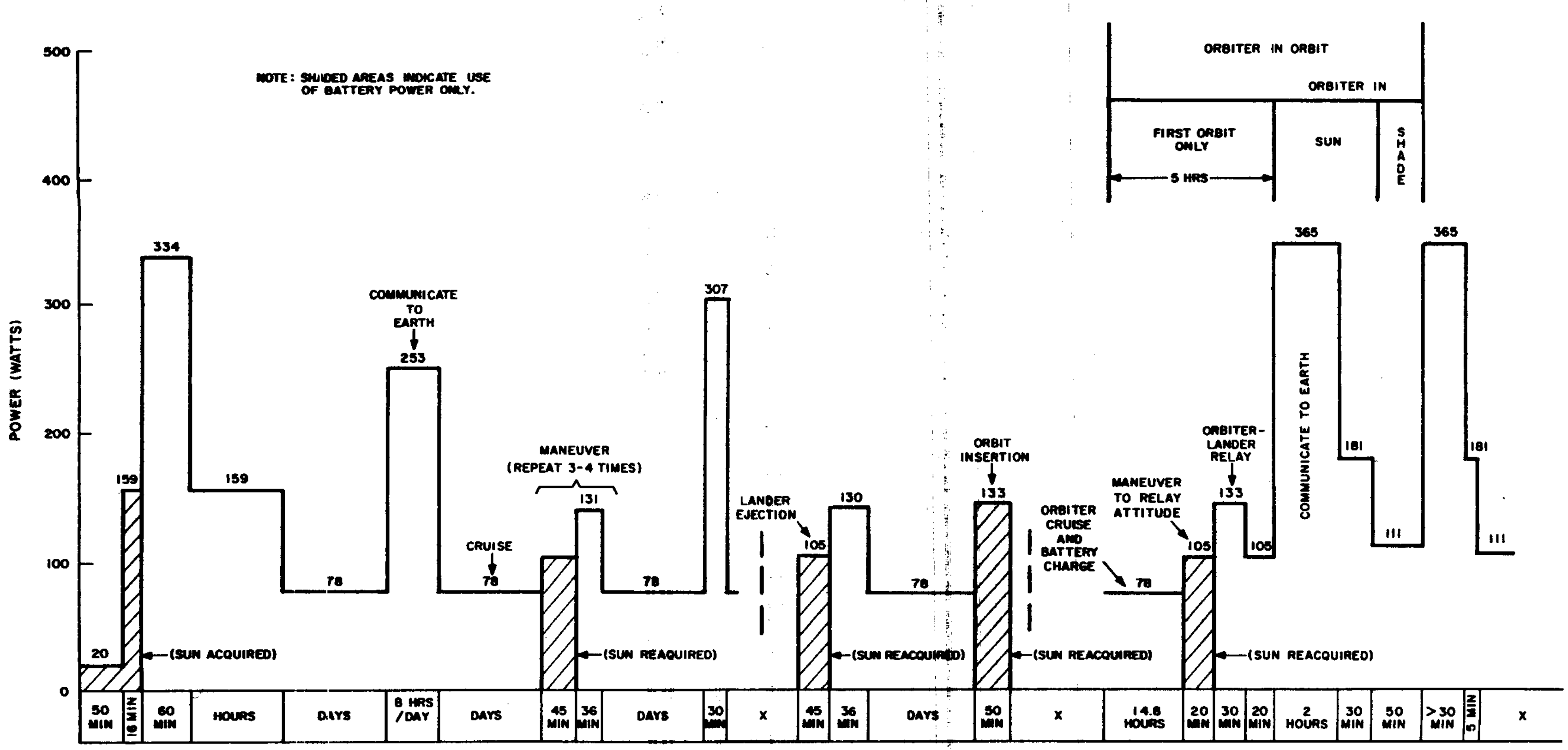
This larger Lander with its capability of surviving on the surface for a minimum of 6.5 hours can perform more sophisticated experiments than were possible in the rather short lived 1970 Venus Lander. There is time to obtain a complete survey of the landscape with the panorama television system. A seismograph can be deployed and a recording of the seismic activity for the entire 6.5 hour life and extensive determination of characteristics of the Venusian surface can be performed. The communication system of the Lander is only equipped with VHF transmitter and receiver. A receiver is provided even though the opportunity for modifying the operation of the Lander is severely limited due to the short life of the Lander.

3.9.3 SYSTEM WEIGHT

The Venus 1972 System weights are given in Table 3.9.3-1.

TABLE 3.9,3-1. SUBSYSTEM WEIGHTS

Sub-System Weights	Venus	1972
Structure		405.98
Orbiter Structure	341.45	
Hardware	42.53	
PHP Structure	20.00	
Hardware	2.00	
Harnessing (Veh.)		78.76
Power Supply		131.20
Batteries	25.00	
Electronics	11.75	
Harness (solar array)	2.89	
Fixed Array	91.56	
Guidance & Control		231.05
Electronics	149.15	
Fe 14 TK & Gas	57.50	
Hardware	24.40	
Communications		248.15
Electronics	216.15	
Antenna (10 Ft. dish)	32.00	
Diagnostic Instrument		30.00
Thermal Control		87.00
Payload		61.00
Scientific	46.00	
TV	15.00	
Propulsion		530.41
Fuel System	405.00	
Pressurization System	125.41	
Orbiting Weight		1803.55



VENUS 1972 ORBITER POWER PROFILE

TABLE 3.9.3-1. SUBSYSTEM WEIGHTS (Continued)

Sub-System Weights	Venus 1972
Orbit Insertion Fuel	2725.45
Lander Weight (one)	2600.00
Midcourse Fuel	221.00
Total	7350.00

The Orbiter structure common to previous missions is utilized here with the modification of reinforcing the upper deck of the Orbiter structure to handle the loads imposed by a single centrally located Lander. A PHP is again provided. Harnessing is fairly extensive and results in an estimate of 79 lbs.

The data requirements for this mission are rather low because there is no orbiting radar and no extensive orbiting television system. Therefore, the 35 watt klystron provides adequate data rate for this mission. A power supply based on solar array and secondary batteries provides a peak output of 365 watts. The battery is sized by the Orbiter requirements during orbit insertion attitude maneuver. The bottom face of the Orbiter provides sufficient area for the solar cell array and therefore the fixed extensions of the Orbiter structure are not required for additional array area. Guidance and control subsystem has the same weight as in the Venus 1970 mission. Diagnostic instrumentation is estimated at 30 lbs. and thermal control at 87 lbs., the same as in previous fully equipped systems. Payload consists of a 15 lb. television system and 46 lbs. of orbital scientific instruments. A 1000 x 7300 nm orbit is utilized in order to provide the weight carrying capacity to accommodate the 2600 lb. Lander.

3.10 VOYAGER SYSTEMS FOR THE TITAN III-C

3.10.1 TITAN III-C DESIGN FEATURES

The Titan III-C is essentially the Titan II with the addition of the "Transtage" as a third stage and the use of two strap-on solids forming "stage zero." The successful flights of Titan II have been augmented with successful full-duration tests of both the stage zero solids and the Transtage in July of this year. The Titan III-C is conservatively designed and a reliability in excess of 0.95 per stage is anticipated. This prognosis is due to the storable propellants, the expected straightforward structural modifications required for the first and second stages, and the Delta, Able-Star, and Agena experience applicable to the design of the Transtage. The first flight of Titan III-C is scheduled for 1965. Potential problem areas exist, however, in proving the flight reliability of segmented engines. The stage zero engines must provide relatively long burning time with no uneven erosion between the two engines and the state-of-the-art is being advanced in thrust vector control.

The Titan III launch complex is designed to the ITL (integrate, transfer, launch) concept which provides rapid turnaround time for a quick succession of launches from a few pads. In this concept, the complete booster-payload combination is assembled and checked out in a vertical integration building, and transported to the pad in a "go" condition except for fueling. This approach permits the integration and checkout of several booster-payload systems simultaneously and up to 20 launches per pad per year. The construction of two pads and a vertical integration complex to handle four booster-payload combinations is now in progress. The cost per launch of a Titan III-C, has been estimated as low as $10 million.

The attitude control system of Titan III-C is flexible and readily adapted to planetary missions. The first stage uses two gimballed nozzles; the second stage uses a single nozzle for pitch and yaw control and a separate small pivoted nozzle for roll control. The Transtage has a complete separate attitude control nozzle system and restart capability.

The features described above for the Titan III-C system and a nominal planetary payload capability of 3400 pounds merit its consideration for Voyager systems. The principal characteristics are listed in comparison with the Saturn 1B + S-VI stage and the Saturn V in Table 3.10.1-1.

3.10.2 VOYAGER SYSTEMS

A study of possible Titan III-C Voyager missions for Mars 1969 showed that insufficient payload capability was available to accomplish a significant portion of the scientific objectives with a system consisting of both an Orbiter and Lander module as was possible with the Saturn C-1B launch vehicle (Table 3.10.2-1). The payload capability is not commensurate with the investment is booster and spacecraft development effort. However, in view of the relative economy of Titan III-C augmented by its launch and checkout features it is possible to launch the different modules with separate launch vehicles: one launch vehicle could booster an all Orbiter spacecraft while another could launch a Lander with its bus. Using this approach, the Titan III-C could accomplish the same mission as a single combined Orbiter and Lander spacecraft launched with a Saturn C-1B.

It should be recognized at this point that the accuracy of the system weight schedule shown for Titan III-C launch systems are not as precise as those provided for the

TABLE 3.10.1-1. COMPARISON OF POSSIBLE VOYAGER BOOSTERS

Characteristics	Titan III-C	Saturn 1B + S-VI	Saturn V
Zero Stage	P-1 Strap on Solids	------	------
First Stage	Modified Titan II	S-1	S-1C
Second Stage	Modified Titan II	S-IV-B	S-II
Third Stage	Transtage	S-VI	S-IV-B
Planetary Payload Capability	3400 lbs	7030 lbs.	60, 000 lbs.
AMR Launch Facility Status	One Interim Pad Being Completed. Two Pads under construction.	One 1B Pad in Use. One under Construction*	3 Pads under Construction
Relative Cost Per Launch**	10 M	30 M	100 M
Estimated Turn-Around Time, Launches per Pad per Year	20	4	5
Estimated Lead Time to Order Operational Vehicles	2 years	2 years	2 years
Expected Operational Date	1966	1966 (Saturn 1B)	1968
Propellants	Solids, N_2O_4 + UDMH + Hydrazine	Kerosene + O_2, O_2 + H_2	Kerosene + O_2, O_2 + H_2

*Handling of S-VI staged Saturn by 1B facilities assumed.
**Costs are relative order of magnitude only.

Saturn C-1B Voyager systems. The attractiveness of the Titan III-C booster vehicle as a Launcher for a Voyager Spacecraft appeared rather late in the study and since the prime purpose of the study was to determine the aspects of a 7000 pound spacecraft for the Voyager system, the Titan III-C study was not as extensive as the spacecraft based on the Saturn C-1B.

Table 3.10.2-1 shows that the estimated payload for the Saturn C-1B launched Mars 1969 Orbiter could be exceeded by an all Orbiter launch with a Titan III-C. However, the best Lander combination would deliver less Lander weight to the surface of Mars than the Saturn C-1B combined Orbiter-Lander.

In arriving at the estimated system performance for the Titan III-C systems it was assumed that the different systems given in Tables 3.10.2-1 had the same thirty day

TABLE 3.10.2-1. TITAN III-C VOYAGER SYSTEMS

Mission	Mars 1969				Venus 1970	
Booster	Titan III			Saturn C1B	Titan III	Saturn C1B
Spacecraft	Orbiter	Bus - Lander	Orbiter & Lander	Orbiter & Lander	Orbiter	Orbiter & Lander
Structure	234	206	220	419	198	415
Harness	70	60	90	106	60	77
Power Supply	218	49	100	218	136	202
Guidance & Control	216	200	226	226	251	231
Communi-cations	276	10	140	291	216	227
Diagnostic Instrumen-tation	30	30	30	30	30	30
Thermal Control	70	10	45	87	87	87
Propulsion	370	115	240	467	443	739
Orbiter Payload	223	—	66	215	107	137
Orbiting Weight	1707	(bus) 680	1157	2059	1528	2145
Lander Weight	—	2230	1030	(2) 1450	—	525
Lander Payload	—	(330)	(80)	(155) (each)	—	(60)
Fuel	1643	90	1163	2072	1977	4590
Total Weight	3350	3000	3350	7030	3470	7260
Orbit (x 1000 n. mi)	1 x 19	—	1 x 19	1 x 19	1 x 15	1 x 4.3

The Titan III-C systems studies evolved a bus-Lander configuration with the bus delivering a single Lander. Side by side packaging of two Landers as in the Saturn C-1B spacecraft was not practical within the Titan III-C shroud volume. In addition, packaging of two Landers vertically in a delivery bus was avoided in order to preclude both the weight penalty of structural load path members supporting the uppermost Lander, and the reliability penalty induced by a separation failure of the top Lander trapping the second Lander. The incentive to consider two Landers in this mission was subsequently increased when reliability analysis of the Saturn C-1B system showed that attainable mission value is increased by using two Landers whenever the total Lander weight exceeds 1840 pounds, which is less than the 2230 pound Lander weight available in the Titan III-C system for MARS 1969. Therefore, any further work on the Titan III-C systems should include an evaluation of two versus one Lander. One advantage that a single 2230 pound Lander will have over multiple Landers is its size permits a larger high gain direct link antenna than was possible with the 1450 pound Mars 1969 Landers on the Saturn C-1B system. Such an antenna with 26 db of gain compared to 20.7 db for the 1969 standard 1450 lb. Mars Lander design provides a direct link communication data rate greater than that achieved in 1969 design. This means that adequate data return from the Titan III-C Lander is less dependent on the capacity of the relay link between the Lander and the Orbiter. The combination, however, provides a much higher probability of successful data return.

Having determined the performance of the various Titan III-C systems for the Mars 1969 opportunity, the question that must be resolved is whether or not the Voyager missions can be accomplished with this launch vehicle in the higher energy years. Consequently, the evolution of the Titan III-C systems out to 1979 was estimated (Table 3.10.2-2). These results indicate that it is possible to perform the Voyager type mission even out to 1979 with the two Lander weights of either 2230 pounds or 2000 pounds. An Orbiter mission was considered only for the first three opportunities which is consistent with the evolutionary program for the Saturn launch system given in Table 3.1-1. However, these Titan launched Orbiters are quite attractive not only from the standpoint of the higher payload mentioned earlier, but from the fact that a circular orbit with the same payload is possible in 1971, which would permit complete coverage of the planet during one opportunity. In addition, if a mapping missions is desired in 1973, it can be performed with a less eccentric orbit then is possible with the Saturn C-1B system.

TABLE 3.10.2-2. MARS TITAN III SYSTEMS

	1969	1971	1973	1975	1977	1979
All Lander System						
Weight Injected, lbs	3000	3000**	2750	2750*	3000*	3000
(Lander Weight	2230	2230	2000	2000	2230	2230
Trip Time (Days)	275	128	167	~325	(less than 1975)	~180
All Orbiter System						
Weight Injected	3350	3600	2800	-	-	-
Scientific Payload in Orbiter	223	223	223			
Orbit, n.mi.	1000 x 19,000	1000 x 1000	1000 x 13,000			

* Type II trajectories but higher than minimum energy trip.

** Higher than minimum energy trip to minimize changes in lander size.

In addition to the Mars missions the use of the Titan III-C was also considered for a Venus 1970 radar mapping mission. The results given in Table 3.10.2-1 indicate that such a mission could be performed with the same mapping radar that was considered for the Saturn C-1B system. However, a more eccentric orbit must be utilized to remain within the capability of the Titan III-C. This increase in eccentricity will reduce the map coverage from approximately 25% to 30% since the radar system is not effective about altitudes of 2500 n.mi. (This difference in coverage will be greater if the two systems were designed for the small life. The Titan III-C mapper life was extended to 225 days to obtain this coverage).

3.10.3 SYSTEM DESIGN ANALYSIS

A. Mars Orbiter System

The structural weight of 234 pounds is estimated on the basis of a compact semi-monocoque structure without Lander provisions, but with the same planet horizontal package incorporated in the Saturn C-1B system. The structure is more compact than the Saturn C-1B Orbiter because the orbit insertion and midcourse fuel requirements are less. The harness weight is estimated at 70 pounds as compared to the (C-1B) 106 pounds, because of the more compact vehicle and the absence of a Lander. Power supply weight is estimated at the same 218 pounds, but the array will not provide the same solar power as in the C-1B Orbiter because the compact structure has less available area for body mounted cells. The guidance and control system weight is estimated at 216 pounds, 20 pounds less than Saturn C-1B, because of the reduced attitude control impulse demands of a more compact vehicle without Landers. The communications system is the same except that only one thermoplastic recorder is used instead of two. This accounts for the 25 pounds reduction in weight. Diagnostic telemetry is the same for all systems. Thermal control is reduced from 87 pounds to 70 pounds because of the smaller fuel tanks.

The total orbiting weight, 1707 pounds permits the same mission to be accomplished as the 2059 pound orbiting weight for the Saturn C-1B system because of the weight penalty induced by the required structure to support the two Landers. This additional structural weight also affects the propulsion system and orbit insertion fuel requirements. Midcourse fuel for a Titan III-C system is only 100 pounds because the total injected weight of the vehicle is 3050 pounds compared to 210 pounds midcourse fuel for a 7050 pound vehicle, for the Saturn C-1B system. The Titan III-C system has a slightly higher payload capability of 223 pounds compared to 215 for the Saturn C-1B. The relay link capability of these systems are the same. Since the orbit is the same, 1 x 19 n.mi. the percentage of television map obtained in the mission is the same as for the Saturn system.

It is noted here that the emergency power supply obtained in the Saturn C-1B system when the spacecraft is not oriented to the sun, from the Lander power supply carried on board the system, is not available in the Titan III-C Orbiter system because there are no Landers. However, there is a slight gain in reliability because the number of maneuvers requiring changes in attitude is reduced since there is no separation sequence.

B. Mars All Lander System

The structural weight of this system, estimated at 206 pounds, is more compact than the structure for the Titan III-C Orbiter because there is no PHP and because the fuel tanks are rather small since only midcourse correction fuel is carried.

Harness weight is reduced to 60 pounds because there is no orbit equipment to be connected. The power supply is 49 pounds, composed of 28 pounds of batteries and 21 pounds of fixed solar array. These are based on a nominal power profile showing a fairly constant 82 watts load with several peaks up to 200 watts. In this case, power is available from the power supply on board the Lander and of course the secondary batteries in the Lander can also be utilized. Guidance and control system is estimated at 200 pounds. This contains all the equipment used for the transit with the exception of the one Canopus tracker and the PHP package drive electronics and actuators with a reduced quantity of attitude control fuel. The communication system is estimated at only 10 pounds because the S-band equipment on board the Lander is utilized through the hard wire connection between the bus and the Lander. A three-foot dish and associate cabling is provided on board the bus. The thermal control requirements are reduced, because of the small amount of fuel on board, to only 10 pounds. Total fly-by bus weight is estimated at 680 pounds. This allows a Lander weight of 2230 pounds to be carried and with 90 pounds of midcourse fuel the total system weight is 3000 pounds. This Lander weight also includes a solid fuel motor to place the Lander on an impact trajectory.

With separate Orbiter and bus-Lander Titan III-C systems, it is possible that a bus-Lander could arrive at Mars without an Orbiter being available as a relay link. With the exception of a relay link in an Orbiter to monitor the atmospheric entry and descent phases of the Lander operation, the Lander mission is not seriously compromised. This is true because the large Lander size permits the incorporation of the higher gain direct antenna utilized in the 2000 pound Saturn C-1B Landers. This antenna with 20.60 db of gain compared to 20.70 db for the 1969 Mars Lander design provides a higher communication rate. It is estimated that the data rate will permit the Lander to send direct to earth, as would be necessary if there were no Orbiter available, approximately 2300 television frames for the first three months of Lander operation. This is approximately two times more than the direct link is capable of in the standard 1969 Lander design because of the gain from the larger antenna. In addition to the higher communication data rate possible in this 2230 pound Lander there would of course be room for additional payload which was not analyzed in detail.

C. Mars Orbiter Plus Lander System

Structural weight is estimated at 220 pounds based on a compact semi-monocoque structure carrying a single small Lander directly above within the confines of a Titan III-C shroud. Harness weight at 90 pounds is just slightly less than the harness weight for the Saturn C-1B system. Power supply is estimated at 100 pounds which includes 60.8 sq. ft. of solar array and 22 pounds of battery. It is estimated that this system will handle a peak load of 346 watts for one hour. Guidance and control subsystem weight is estimated at 226 pounds, exactly the same as for the Saturn C-1B system. Thermal control is lower in weight because of the small quantity of orbit insertion fuel.

With the same orbit (1000 x 19,000 n.mi.) that was chosen for the Saturn C-1B the orbiting payload is only 66 pounds while the Lander payload of only 80 pounds is also much smaller than the Saturn C-1B system. The amount of payload that can be carried in this mission is a much smaller fraction of the total injected weight than for the separate Orbiter and bus-Lander missions described above. There are serious packaging and configuration problems that are described in more detail in Section 2 of Volume IV. Although a television map of the planet can be obtained with this small Orbiter, the total television information that can be returned is quite small because of the requirement of operating a communication system for only a total of four hours per orbit. This somewhat marginal system is not recommended for the Mars 1969 Voyager system.

D. Venus 1970 Orbiter System

Since the prime mission for a Venus Orbiter is the acquisition of a radar map of the surface of the planet, the Orbiter structure for this mission is envisioned to be an extremely compact one with a single hinged package consisting of a 10-foot antenna with a small package of instruments mounted axially with the antenna, instead of the separate planet horizontal package, radar antenna and the orientable high-gain communication antenna on the Saturn C-1B radar Orbiter. The harness is estimated at 60 pounds because of the compactness of the vehicle and the one antenna. The orbit for this mission is by necessity highly eccentric, i.e., a 1000 n.mi. periapsis and a 15,000 n.mi. apoapsis. With an orbital period of approximately nine hours the surface of the planet progresses 32 n.mi. per orbit underneath the orbit plane. With a swathwidth on the radar scan of 112 n.mi. the radar need only obtain one complete swath for every four orbits. The time during which the Orbiter is below the effective operating altitude of the radar, 2500 n.mi., will be approximately 30 minutes. One quarter of this time, 7.5 minutes, is the amount of time that the radar has to operate per orbit. Therefore, dark time power requirements plus the radar system power require 20.6 pounds of batteries. The radar data can be handled with a communication rate of 1.2 kilobits per second with a total operating time of 8 hours per orbit, at a power level of ten watts. With a 20% overall estimated efficiency, the power required for a communication system is 50 watts. Housekeeping power plus communication input power plus power to recharge the battery for the radar and dark time operation results in the estimated solar array requirement of 204 watts. Estimated weight of array plus required hardware and regulators and the batteries is 136 pounds.

Guidance and control system is estimated at 251 pounds. This is caused by increased allowance for attitude control gas supply. The high eccentricity of the orbit and long period only permit one sided viewing by the radar system. Therefore, to increase the mapping coverage, the operational life was set at an entire Venusian year of 225 days. The communications estimated weight is the same as for the Venus 1970 Saturn C-1B system with the exception that a separate dish is not provided. After the radar information is obtained the radar dish must be oriented toward earth, every orbit so that the acquired radar data can be transmitted.

The payload estimated at 107 pounds includes the radar equipment at 57 pounds, the radar dish at 32 pounds plus an additional allowance for scientific instruments and additional structure to hold the instruments coaxial with the antenna.

The capability of this Venus radar Orbiter is somewhat less than the Orbiter carried on the Saturn C-1B 1970 system. However, a radar map of approximately 30% of the planet can be obtained with a long stay time in orbit of 225 days.

The configurations of the Titan III-C Voyager systems are given in Figure 2.3.5-1, and 2.3.5-2 of Volume IV.

3.10.4 COST COMPARISON

In order to obtain a meaningful first approximation as to the relative cost of a Saturn C-1B and a Titan III-C Voyager system it is necessary to place the two systems on a comparable basis in terms of mission capability. Considering the Mars 1969 mission it appears that two Saturn C-1B systems would be somewhat comparable to five Titan III-C launches in terms of mission capability. The two Saturn launches represent two Orbiters and four Landers (1450 pounds) whereas the five Titans represent two Orbiters and four somewhat smaller Landers (1100 pounds) plus one large Lander (2200 pounds). Since the total payload weight in the five Titans is greater than the payload in the Saturn systems any cost comparison must take into account this increased capability. Based upon a judgment as to the added value of the increased payload, the Titan III-C costs were multiplied by a factor of 0.835 to place the two systems on a comparable basis.

Another difference between the two systems is the cost of the spacecraft. Since the Titan III-C systems employ a greater variety and number of modules, there should be an increase in development and production cost. However, since detailed cost estimates of the Titan III-C systems were not made, a range of values were assumed to bracket the possibilities.

The resulting comparison is given in Figure 3.10.4-1. This figure shows a comparison which also includes the effect of the depreciation of the development costs of the S-VI stage, (estimated at 200 million) over the eight Voyager launches. Since a range of values were obtained for the unit cost of the launch vehicles, the comparison is given as a function of these costs with the most probability limits so indicated. The eventual costs of each vehicle would probably fall somewhete within this square. Actual Titan III-C and Saturn C-1B unit costs will locate a particular point on this plot. If this point is above the shaded band, then the Saturn C-1B is the more economical launch vehicle and conversely Titan III-C is favored below the band. If S-VI stage costs are charged to the eight Voyager flights, then the probability is very high that Titan III-C will be the more economical vehicle. With no development costs charged to Voyager a more precise costs analysis must be undertaken to determine which is the more economical system.

3.10.5 CONCLUSIONS

It is concluded that the Titan III-C is an attractive launch vehicle for the Voyager mission and warrants more detailed consideration. The study results show that it can perform the same missions as the Saturn C-1B without any degradation of capability (with the single exception of the radar mapping of Venus) at a reduced cost especially if the S-VI stage development were charged entirely to the Voyager program. In addition, the development of the Titan III-C launch vehicle is now underway, while the S-VI stage required for the Saturn C-1B is only in the conceptual design stage.

Titan III-C systems utilizing bus-Lander configurations with delivery of two Landers from a single bus should be studied to determine merits of the system in comparison with single Lander configurations. In addition, it is suggested that a detailed cost effectiveness study should be completed to accurately compare Titan III-C and Saturn C-1B.

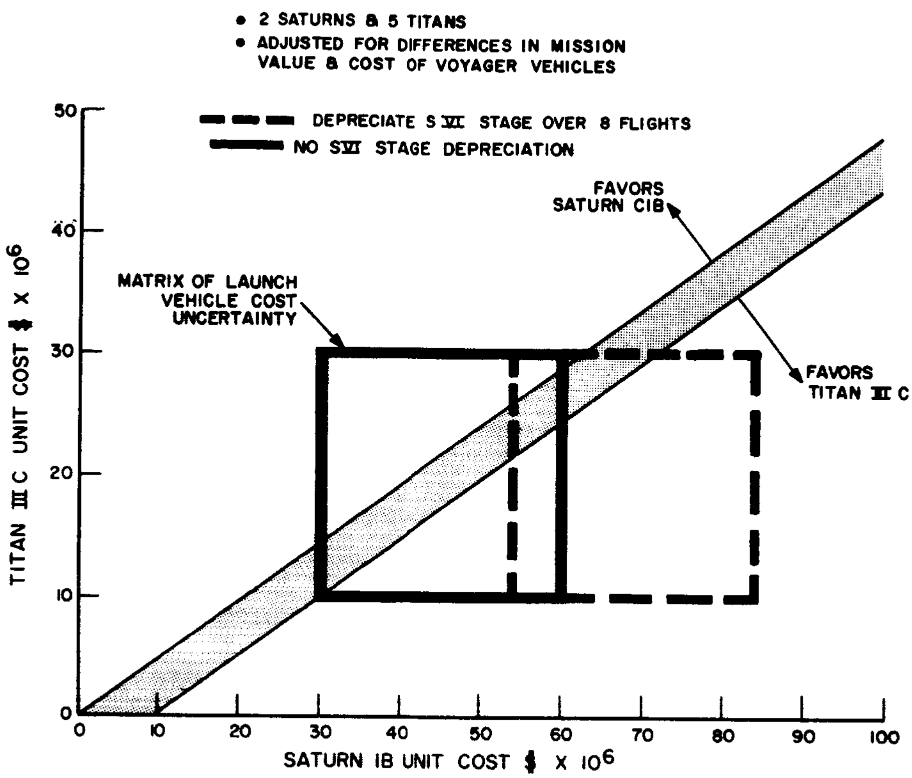


Figure 3.10.4-1. Cost Comparison of Saturn I-B and Titan III-C

3.11 SPACECRAFT GROWTH CAPABILITY TO 60,000 POUNDS

In the previous section, an analysis was presented which indicated the possibilities of fulfilling the Voyager objectives with a spacecraft in the 3500 lb. class (Titan III-C). The results indicated that this is an attractive system employing the same type of landers and orbiters that were employed in the 7000 lb. class system with, for the most part, identical subsystems. However, the question of employing design compromises in either the 3500 lb. or the 7000 lb. class vehicle to accommodate growth to a 60,000 lb. spacecraft (compatible with a Saturn 5 launch vehicle) is a difficult one to answer since this represents almost an order of magnitude increase in size.

The question is further complicated by the fact that the results given in this report indicate that there is no requirement for such weight capability. The 7000 lb. or the 3500 lb. class spacecraft can provide a vastly greater amount of scientific information about Mars and Venus than the Mariner B and can also provide the information needed to design and manned Mars landing system. No payload could be identified that would require the Saturn 5 capability. In other words the next step after the presently proposed class of Voyager spacecraft would be a Manned Landing System.

It is realized, however, that based upon some unexpected results from early Voyager systems or if a manned Mars landing mission were delayed much beyond the next twenty years, a requirement may be established for a vehicle in the 60,000 lb. class. However, at this time there is no satisfactory means of determining what such a spacecraft might consist of to perform a mission that is not identifiable.

With these reservations, a number of system possibilities were hypothesized in order to determine if these systems could be evolved from the 7000 lb. spacecraft.

Based upon the results of the Voyager studies it is possible to set forth some desirable characteristics that may be required of a 60,000 lb. spacecraft. One of these would be the use of a more sophisticated guidance system in order to increase the accuracy with which a lander could be placed on the surface. This may be necessary if a rather small area is identified which would warrant a more detailed investigation. Another design characteristic would be a lower "g" environment. With the large landers and the associated greater payload capacity of the Saturn 5, some of the payload components will most probably be more "g" sensitive than those employed in Voyager. Therefore, the surface impact system would be designed for a "g" level much lower than the 125 g's for Voyager. The entry "g" level will not be a problem since the entry angle must be low (20-28°) because of ballistic coefficient restrictions.

By the time the 60,000 lbs. spacecraft may be employed, the Mars atmosphere will be fairly accurately defined. Therefore, it may be possible to use atmospheric braking for an orbiter with the associated savings in propulsion system weight. This, however, requires a tight control over the entry corridor which will necessitate a more sophisticated guidance system. However, as it will be shown in later paragraphs, the propulsion weight that is saved by use of this technique will not permit the spacecraft to carry any greater weight in the lander (or greater number of landers) since the Saturn 5 is volume and size limited (shroud) rather than weight limited. The shroud diameter on the Saturn 5 in the same size that is postulated for the S-VI stage on the Saturn C-1B. This limitation will restrict the weight of the landers since a certain value of ballistic coefficient must be exceeded if satisfactory parachute deployment is to be achieved or if retro rockets are to be fired at the proper velocity and height above the surface.

Based on the 11 mb model atmosphere, an estimate was made of the maximum allowable lander weights for a number of system possibilities and guidance accuracies (Table 3.11-1). The results indicate that landers will be restricted to weights between 9000 and 20,000 lbs. depending upon the degree of sophistication employed in the lander and guidance system.

It is evident that in order to maximize the weight landed on the surface of Mars a parachute system should be employed. The use of retro rockets (even in combination with a parachute) is not attractive especially on a weight basis. This conclusion sion applies only to the use of retro rockets prior to parachute deployment or as the sole retardation system. These large landers will still employ a small retro system after parachute deployment but just prior to impact since studies have indicated that a minimum weight for a final retardation system is obtained with a combination of parachutes and retro rockets for any reasonable impact "g" level - $<$ 125 g.

With these lander weights it is possible to hypothesize a number of system concepts such as those given in Table 3.11-2. The ones chosen represent three of the more attractive concepts. Because of the shroud length of 56 feet only two landers could be employed. Therefore, to utilize the total 60,000 lb. capacity a larger orbiter was considered on two of the concepts, whereas for the third, the landers were considered to be released from orbit (Some design details on Concept A are given in Section 2.3 of Volume IV).

Reviewing these concepts it is evident that the landing and orbiting modules for the Voyager system could not be considered to grow to the 60,000 lb. spacecraft except on perhaps a subsystem basis.

The three subsystems in the Voyager systems that show the greatest promise of growth potential are power supply, communications and guidance.

3.11.1 POWER SUPPLY

The 70 watt radioisotope generators being planned for Voyater would certainly have utility in the larger 12,000 lb. landers. It may pay off in terms of reliability to use these units in multiples (depending upon power requirements) rather than develop a single larger unit.

3.11.2 COMMUNICATIONS

The high data rate (16,000 bits/sec) capability of the Voyager communications system would probably be utilized directly for the 60,000 lb. class spacecraft since it is difficult to foresee any requirement for a higher rate before a manned system.

3.11.3 GUIDANCE

The Voyager guidance system has the required accuracy if the landers are released from orbit. However, if further analysis indicates that a direct entry is desirable, the accuracy of the system must be improved. The present guidance system can grow to the accuracy by the addition of a radar giving planet range information and tightening up on the requirements on a number of the other components.

TABLE 3.11-1. MAXIMUM LANDER WEIGHTS*

(11 mb - Atmosphere)

	Systems	Voyager Guidance Accuracy ± 4° Entry Angle	Improved Guidance Accuracy ± 2° Entry Angle
I	Direct Entry - M 2.5 Parachute	9,000-10,600** lbs	11,000-13,000
II	Direct Entry - M 5.0 Parachute	11,600-13,700	17,000-20,000
III	Direct Entry - M 2.5 Parachute + Retro from 21,000 to 15,000 ft/sec	9,200-10,800	11,000-13,000
IV	Direct Entry - M 2.5 Parachute + Retro from M 5.0 to 2.5	8,200- 9,700	12,000-14,000
V	Lander Released from Orbit - M 2.5 Parachute	12,400-14,600	-
VI	Direct Entry - Retro from M 5.0 to 0	6,400- 7,000	9,400-11,000

* The lander weight values are given for the condition after retro firing but before parachute deployment. Retro rocket hardware is considered included in the weights.

** The two values represent, respectively, the maximum weight (based upon the maximum drag coefficient and a 260" diameter) for a ballistic, aerodynamically stable vehicle shape and the maximum weight for a ballistic vehicle requiring some other form of attitude control.

Note: Minimum acceptable altitude for parachute deployment:

M 2.5 - 20,000 ft

M 5.0 - 25,000 ft

As mentioned elsewhere the present Voyager study was based upon the use of the Saturn C-1B + S-VI stage and the Titan III-C. No detailed design analysis was undertaken on a spacecraft for the Saturn 5 (other than that discussed in the previous paragraphs) since the present study indicates that a spacecraft based upon either the Saturn C-1B or the Titan III-C would make an extremely attractive system. However, if a number of developmental Saturn 5 flights were to be made in early 1969 without a presently assigned payload it may be desirable to place one or two of the presently designed Mars 1969 Voyager systems on this booster to obtain additional flights. (Since the full payload capacity of the Saturn 5 would not be utilized, a larger launch window and/or faster trips could be employed.)

Another possibility (attractive for budgetary reasons) would be to plan the Mars 1969 flights with developmental Saturn 5 launches (uses one or two Mars 1969 Voyager systems) and delay the Saturn C-1B plus S-VI Voyager launches until 1971.

TABLE 3.11-2. 60,000 LB. SPACECRAFT CONCEPTS

(MARS only)

	(A) Direct Entry	(B) Lander Released from Orbit	(C) Direct Entry	(D) Direct Entry
	Improved Guidance M 2.5 Parachute System	Voyager Guidance Accuracy - M 2.5 Parachute System	Improved Guidance M 5.0 Parachute System	Improved Guidance Retro M 5.0 → 0
2 Landers -	12,000 lbs. (each)	13,000 lbs. (each)	18,000 lbs. (each)	10,000 lbs. (each)
Lander Retro	Fuel	-	-	8,000 lbs. (each)
Orbiter	12,000 lbs. (1000 x 1000 n. mi. orbit)	4,000 lbs. (1000 x 19,000 n. mi. orbit)	11,600 lbs. (1000 x 19,000 n. mi. orbit)	11,600 lbs. (1000 x 19,000 n. mi. orbit)
Fuel	24,000 lbs.	29,000 lbs.	12,400 lbs.	12,400 lbs.
TOTAL	60,000 lbs.	60,000 lbs.	60,000 lbs.	60,000 lbs.

3.12 AREAS REQUIRING ADDITIONAL STUDY

The results of the analysis on the Voyager system suggest that additional refinement in a number of areas is desirable in order to improve the overall system capability. Some of the more important ones are given in the following paragraphs.

1. Continuous low power transmission in the Lander direct link should be considered. This would cause an increase in the size of the RTG but initial figures indicate that the total data transmitted per day would be higher than the present intermittent system.
2. The trade-off study of high gain antenna pointing requirements versus data rate, antenna diameter, and communication power supply weights, especially in the case of the continuous low power transmission, on both the Lander and the Orbiter should be extended. This would improve the definition of attitude system control requirements.
3. A more careful analysis should be made of low data rate communications systems for missions other than Mars 1969 and Venus 1970. Here the trade-off is between the reduced development costs due to common use of one system power level for many missions and the savings in power supply weight and mission weight for systems sized to satisfy mission requirements.
4. The effect on the spacecraft operation of tape recorders of the conventional reel type should be determined if the thermoplastic recorders are not available for the high volume information recording requirements for the Mars 1969 and Venus 1970 missions. Mechanical tape recorders have less flexibility and might possibly affect the attitude control requirements of the vehicle.
5. Investigate the application of increased or variable focal length optics on nadir vidicons to reduce the effect of orbit eccentricity on resolution obtained in the TV mapping missions for Mars 1969 and 1971.
6. The advisability of incorporating a medium gain parabolic antenna fixed to the Orbiter body should be investigated. This antenna would either be a substitute for the orientable high gain antenna or, as a back-up, would be operated in a mode where the Orbiter itself must be slewed around to aim the antenna at the earth; this would provide adequate data rates for some television returns to earth if the pointing mechanisms on the high gain antenna should fail.
7. The effect of an all radiosotope thermoelectric generator power supply on the Orbiter operation and design should be appraised if the availability of radiosotopes improves or the launch dates slip beyond 1969.

SECTION NO. 4. RELIABILITY

4.1 INTRODUCTION

The distinct possibility that the Voyager Program will provide man's first opportunity to obtain detailed information from the surface of nearby planets and that such knowledge will be of great significance in adding to our understanding not only of the larger aspects of the interplanetary environment in which we live, but also of life itself, make its successful initiation and completion most important.

The very infrequent and limited opportunities for the launching of such a space system (i.e. once every two years, or so) and the very high increases in the overall program costs which would result from the 2 year slippage increments involved in failure to obtain successful results during a given year's opportunity, make it essential that every practicable effort be made to avoid these extra costs.

These reasons in addition to those generally applicable to manned and unmanned space projects make the reliability of each operational system and element of unusually great importance to this program.

4.1.1 SCOPE

The principal efforts during the study contract have been those related to optimization of system concepts and to the identification and evaluation (i.e., reliability estimations) of alternative subsystems, components and operational plans to establish a quantitative basis for those optimizations and provide a reasonably accurate indication of the attainable system reliability (i.e., probability of mission success).

Directly related to this optimization and reliability analysis have been the efforts contributing to the establishment of an overall development program plan which would implement their attainment. This has included the preparation of key items of high reliability engineering standards so that their cost and schedule effects might be incorporated. Also included has been the preparation of a "Contractor and Subcontractor Reliability Requirements" document, its review and discussion with representatives of principal participating contractors, and its documentation to them to assure that quotations and estimates of cost by all contractors reflect this scope of effort.

The work undertaken by potential subcontractors to assist in the overall system definition and optimization is recorded in their own reports which accompany this final report. This was integrated to a considerable degree by discussions and reviews during the course of the study; however, variations in individual approaches, designs, reliability evaluations, etc. may be evident. It is felt that these separate reports substantiate the conclusions reached and the recommendations made in this final report.

4.1.2 APPROACH

In the early systems "trade-off" analyses, the general knowledge of key engineers as well as quantitative comparisons with similar components and subsystem elements were used to indicate preferred system alternatives. This included the determination of functions and power consumption, weight and environmental stress estimates. Failure densities per watt of dissipation, etc., were established to assist in these early estimates of reliability. It has been of interest to note during the subsequent subsystem and component definition and detailed reliability estimations that good general agreement was found in these early approaches.

Design simplicity together with statistically adequate design margins was and remains the basic approach to obtaining a reliable system. The use of the rotation of the planet Mars in place of an active tracking antenna system to sweep the beam of the high gain antenna over the earth once each day and thus permit the simplification of that antenna's guidance and control system as well as to make it possible to simplify and also to de-activate this antenna control subsystem is illustrative.

Once a workable system was established, its possible simplification was analyzed and reviewed. Its components and modules were also evaluated as to weight, power consumption, and reliability. Proceeding in order of priority of greatest improvement in system reliability per pound of added weight, including the effects of the power requirements upon weight, with those elements and groupings whose function permitted of redundancy, (i.e., providing a back-up mode of operation or secondary unit which could perform the needed function should the first fail to function properly), the system design was made more reliable by the inclusion of backup subsystems until the weight capabilities of the system were exceeded. Again, the re-simplification of approach and the consideration of alternative component designs capable of improved performance, better stability, greater reliability at reduced weight was undertaken. This progressive, iterative process, largely documented by revised block diagrams, layouts and internal memoranda, and to a great degree involving "engineering judgments" and reviews by key engineering personnel and at times by discussions with key personnel of applicable subcontractors, provided the mechanism of design and system evolution.

Similarly the consideration of interplanetary trajectories were evaluated in terms of transit times and weights of propulsion, communications, etc., within the available launch opportunities and safety requirements. Where alternatives offered longer interplanetary transit times but greater weight carrying capacity, specific applications of this weight for reliability improvement were considered together with the adverse effects of longer operational and standby periods.

In the great majority of such analyses in the design or systems concept development, the relative magnitudes of improvement were evident and the documentation of their more exact examination was not needed.

As soon as the block diagrams for the components of the system were made in sufficient detail for a more complete function analysis and component composition to be established (or estimated) at piece part level, more formal reliability estimates were prepared using the best available part and component information and using High Reliability parts (e.g. Minuteman, Advent, etc.) wherever such parts could be considered applicable.

Backup modes of operation were evaluated as well as primary modes. Although system optimization and subsystems design improvements have continued steadily even to the time of preparation of this report, it is felt that this progressive, iterative approach has provided a "best estimate" of the system reliability. It has also been effectively participated in and responded to by the system and subsystems engineers so that a well balanced, soundly based approach to the overall NASA objectives and requirements for the Voyager mission and system is felt to have been provided.

4.1.3 PLANNED OBJECTIVES

As stated in "Section 11 Reliability" of General Electric's Technical Proposal No. N-20053, 25 March 1963, by which this study contract was undertaken, the following outputs were outlined;

1. A mission reliability requirement with appropriate trade-offs, reliability apportionments to system and component level and reliability predictions for the recommended system;

2. System and subsystem component designs incorporating as a result of the failure effects analyses, redundancy, backup modes of operation, fail safe logic, and protection against the hazards of launch environment, interplanetary travel and entry and landing accelerations to insure maximum data return;

3. Specifications for design and testing of systems, subsystems and components that will be required to meet the apportioned reliability;

4. A recommended experimental program to determine the effects of sterilization on the reliability of the parts and subsystems identified during the study;

5. Reliability and quality assurance requirements and programs to be imposed on the hardware contractors;

6. A recommended test program for parts, subsystems and systems to demonstrate performance and reliability in simulated space environments.

Each of these outputs is provided in or referenced (and separately supplied) by this final report. Items 3, 4, 5 and 6 are provided as portions of or appendices to the Reliability and Quality Assurance Program Plan as referenced in Volume VI.

4.2 DESCRIPTION

4.2.1 SUMMARY

A. Program Implications of Reliability Analyses

The results of reliability analyses with the incorporation of redundancy in the various subsystems as definitized by this study report clearly indicated the following:

(1) Practicability

A Voyager System undertaken along these lines is practicable and is expected to be well within the attainable "state-of-the-art" applicable to the program plan as provided in Volume VI of this report.

(2) Mission Success Probability

Mission success will be attainable in at least an average of three out of every four flight opportunities. In this instance, recognizing the contribution of each of the instruments, mission success is defined as the successful return to earth of at least 75% of the scientific value of the many scientific instruments carried. A flight opportunity presumes the successful operation of the launch vehicle.

(3) Meticulous Engineering Efforts are Required

The attainment of these successes will require the thorough and consistent application of (a) design standards, (b) selected materials, parts and processes, (c) safety factors, margins and allowances, (d) extensive testing, screening and evaluations for design development and quality assurance, (e) meticulous attention to detail in all portions of the program, (f) adequate time in the schedule for incorporation of the results of tests and evaluations into the design and production of the actual flight hardware, and (g) the reliability-life testing of final flight hardware components, subsystems and systems to demonstrate and verify that the designs and flight hardware actually launched are of the quality required. These design standards, etc., must be established early in the program and must be of the best quality attainable. All reliability analyses for this study have been based upon this premise and have used the low part failure rates applicable to programs in which such standards were developed and used (e.g., Minuteman, Advent, etc.). Similar reliability analyses based on average military standards have indicated that a probability of success of less than 1 percent is applicable to the Voyager program if so conducted.

(4) Sterilization Effects on Reliability

The inquiry made into the parts, materials and processes which are required for this program has revealed numerous design areas and has identified many specific items which are not sterile as presently manufactured. Also, many show degraded performance if sterilized as required for this program. Nevertheless from the study made to date it is considered that sterilization requirements can be satisfactorily achieved and sterilization (and resterilization) of all Lander components can be performed immediately prior to launch without adversely affecting the attainment of the system reliability and mission success noted above.

(a) Such attainment in practice will be best assured by the immediate or early redesign and development of these components, modules and material formulations and processes which this study (and related efforts under other programs and contractors) shall have identified as not being presently adequate for sterilization by "dry heat" procedures.

(b) It is significant to note the high percentage of the total complement of parts required for a Voyager System which are now fully sterilizable by "dry heat" procedures. Also, that in every instance in which redesign and development effort has been authorized or in which a detailed feasibility study carried out, a suitable solution or method of approach to redesign has been found.

(c) The verification of the performance capability and reliability of all sterilized components and component parts, materials and processes by reliability-life tests is considered essential to the success of this program. Such verification is included separately in the "Voyager Lander Parts Sterilization Compatibility Program".

(5) Terrain Suitability

A review of available information both published and unpublished relative to the nature of the surface environments, atmosphere, etc., of Venus and of Mars has been made in the course of this study. With regard to survival of surface impact and subsequent operation by the Lander(s) on the surface of Mars, the terrain is indicated as being 75 percent desert. The degree to which rugged, rock filled canyons or other terrain may exist which is of such a nature as to present a high probability of serious, permanent damage or entrapment of the Lander(s) is not determinable.

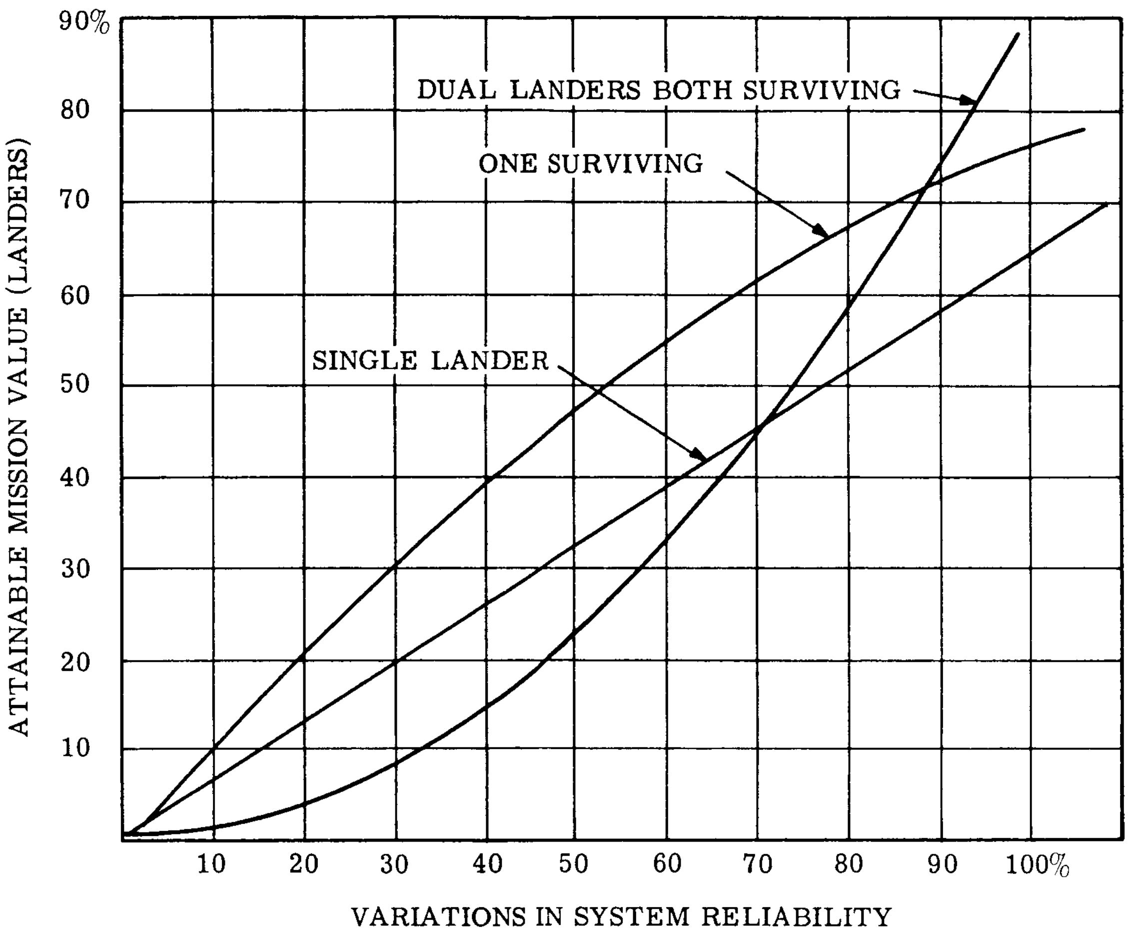
Figure 4.2.1-1 provides graphic illustration of the effect of the super-position of any such variation in overall system reliability.

In the preparation of the data from which these curves were plotted the additional weight available in the single Lander for increased payload (i.e., scientific instruments) was all applied to increase the systems reliability by the inclusion of complete, redundant RTG power supply thermal control, Lander communications, increased depth of impact protecting crush up material, etc. Thus the mission value of the complement of scientific instuments in each of the Landers was both complete and identical. The term "Attainable Mission Value" represents the product (i.e., integral) of the applicable System Reliability and the Value (in percent) of the Lander's scientific instruments as a portion of the total value (100 percent) available with one completely successful Lander together with one completely successful Orbiter.

Thus, Figure 4.2.1-1 provides a direct comparison of the most probable results obtainable with each of the system configurations shown. The vertical line at 90 percent Terrain Reliability (this was considered to be a more than adequate contingency since no more than 10% of the Terrain should be of such adverse nature as to seriously damage a Lander System of the design proposed by this study for the Voyager System proposed) thus provides a 10 percent contingency for the unknowns of Martian terrain. The effects of this or any other contingency (i.e., in Reliability) which further design, development, experiment or analysis might make pertinent to the Voyager could be considered directly on this (or this type of) chart.

It would appear that the results obtainable from the high reliability for at least one Lander surviving from a dual Lander system (curve No. 2) exceeds the value of a single Lander system for all superposed reliability considerations (e.g., terrain, booster, etc., as well as the reliability effects of any design characteristic, revision or reliability calculation applicable to all the Lander configurations studied).

It is also of interest to note that the mission value attainable through the second set of instruments and unique location and environments of the second Lander of a dual Lander System is sufficient to compensate for the higher risks involved (i.e., lower probability) in having both Landers of the dual system survive impact and provide fully satisfactory performance during the first month after arrival on the planet. At a system reliability (including terrain effects) greater than 75 percent of those which have been calculated



NOTE: MARTIAN TERRAIN RELIABILITY ... "T" ... WAS INCLUDED IN ATTAINABLE MISSION VALUE ANALYSES AS ... 90%

4.2.1-1 Attainable Mission Value (One Month) vs Variation in System Reliability as best estimates (i. e., most likely values) for the proposed Voyager dual Lander configuration, this compensation is sufficient to make it, curve No. 1, of greater value than the single Lander configuration, curve No. 3. Also, at a system reliability greater than 88 percent of that applicable to the proposed design, the attainable value with dual Landers - both surviving, curve No. 1, exceeds that for dual Landers - one surviving, curve No. 2.

(6) System Reliability Analyses to resolve the one (1) Lander vs two (2) Landers systems configuration

To quantitatively establish the Attainable Mission Value of which each configuration is capable, each of the scientific instruments was carefully reviewed by the responsible scientists, systems and reliability engineers and a portion of the total value, 100 percent, of the mission's full complement of instruments was apportioned to each instrument. These instrument "available" values and their accrued value at given times after arrival together with detailed methods of analysis are documented and described in section 4. 5. 1.

30 percent of the "Available" mission value/system was assigned to the Orbiter, 10 percent to the Entry data of the first Lander and 60 percent to the Surface data of the first Lander. The Surface data from a second Lander with identical instrumentation, whether from the first system launched or from a subsequent system, was considered to be of equal value, namely 60 percent. However, because the entry data from a second Lander would largely represent a second duplicative set of readings (not being "geographically" unique by reason of its location), this value from a second Lander was reduced (from 10 percent) to 5 percent. The total "available" values are shown in Figure 4. 2. 1-2.

When each instrument value (or value increment over a given time) has been multiplied by its corresponding reliability, ranked in order of greatest attainable value per pound, and applied in that order to the net payload capability of a Lander (and correspondingly for each of 2 Landers) an attainable mission value vs entry/Lander weight is obtained per Figure 4. 2. 1-3. From this type of analysis, including as it does the cumulative effects of all the earlier performance, weight, scientific and reliability trade-offs the decision point (e. g., 1840 lbs total weight available for one or two Landers) is clearly shown. Below this point a one Lander system is advantageous. Where available Lander(s) weight is greater, the marked advantages of dual Landers are shown (approximately 2900 lbs has been shown by the study to be available for Landers to Mars in 1969).

One of the many alternatives studied covered a reduced Lander communications rate capability in which the reduction in power supply plus communications weight was reduced by 100 lb. The effect of this was to reduce the decision point weight from 1840 to 1200 lb.

As total Lander weight is available above the decision point, the relative value of the data from the second Lander may be established at less than equal (i. e., 100 percent) of that of the first Lander with dual Landers still being preferred. Such a systems reliability trade-off is shown in Figure 4. 2. 1-4.

At later opportunities, should surface roving vehicles or other key scientific capabilities be considered to be a major portion of the total mission objective, a single Lander providing the required payload weight capability may be of first importance. Also, if low atmospheric densities (11 millibars or other comparably difficult requirements) should greatly reduce the net payload capability of a dual Lander System, a single Lander System may be preferred. However, for the Voyager Systems design proposed by this final report, and for those studied in arriving at the finally recommended design, two Landers would appear to be definitely advantageous over a single Lander System.

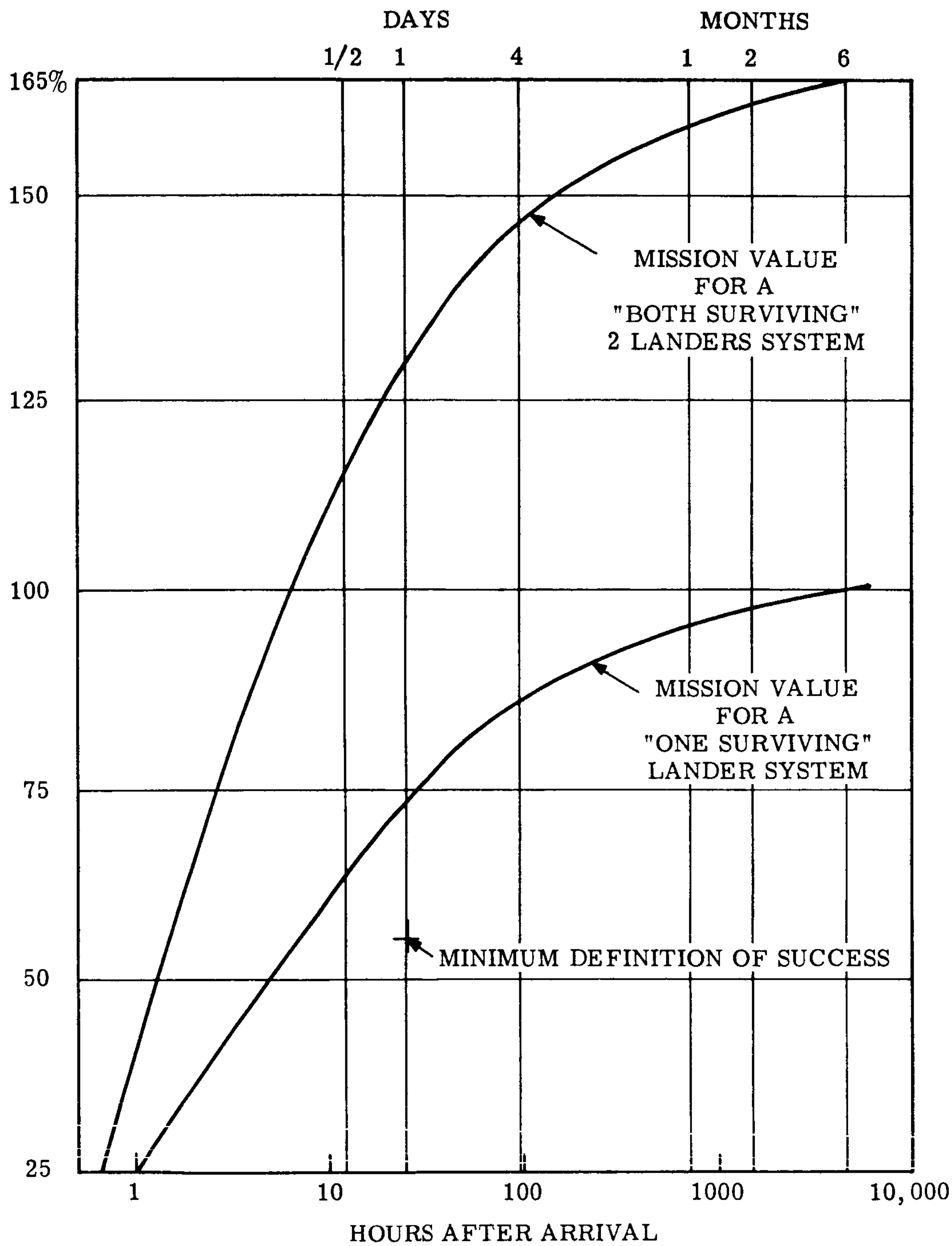


Figure 4.2.1-2 Mission Scientific Value

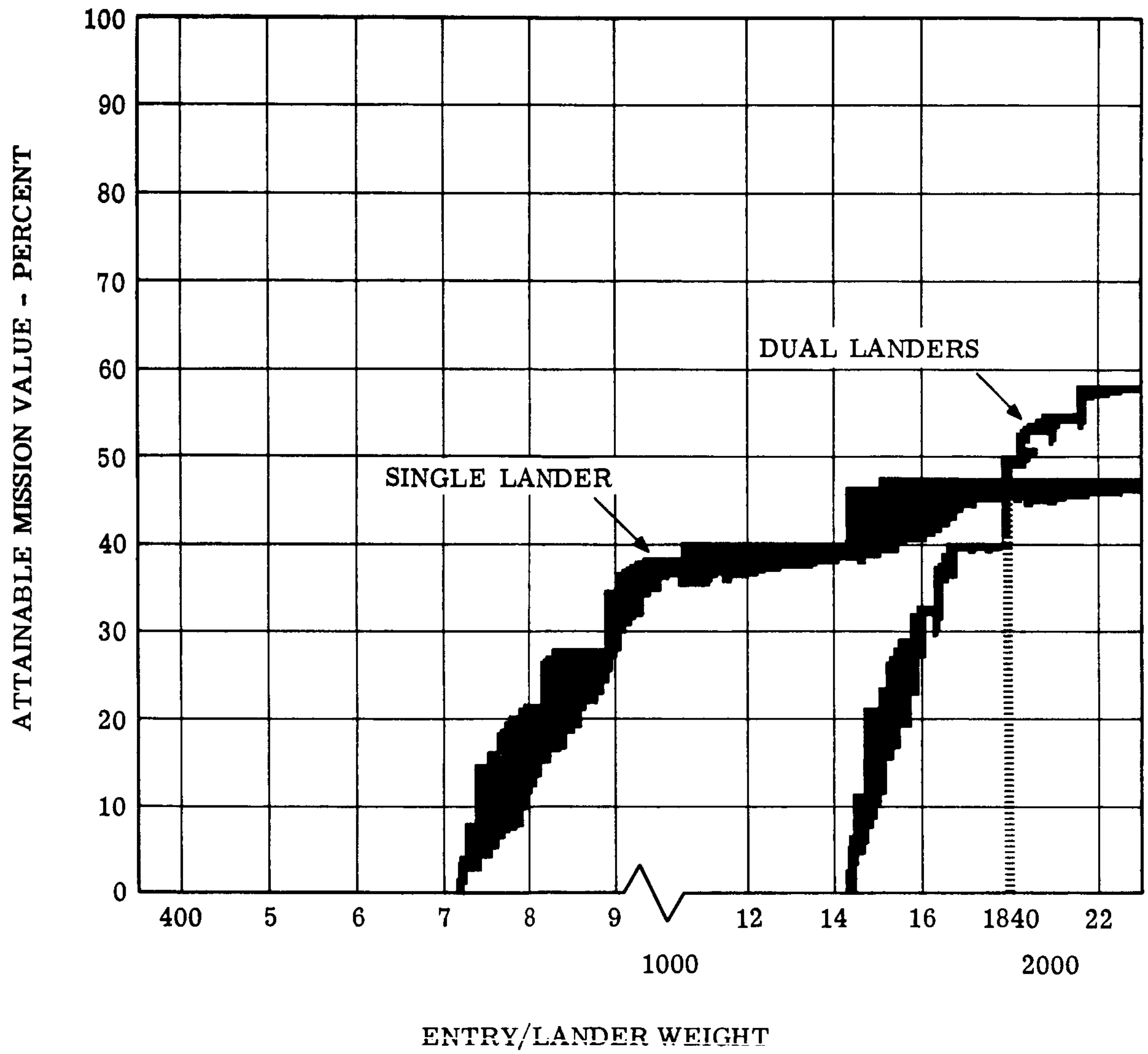
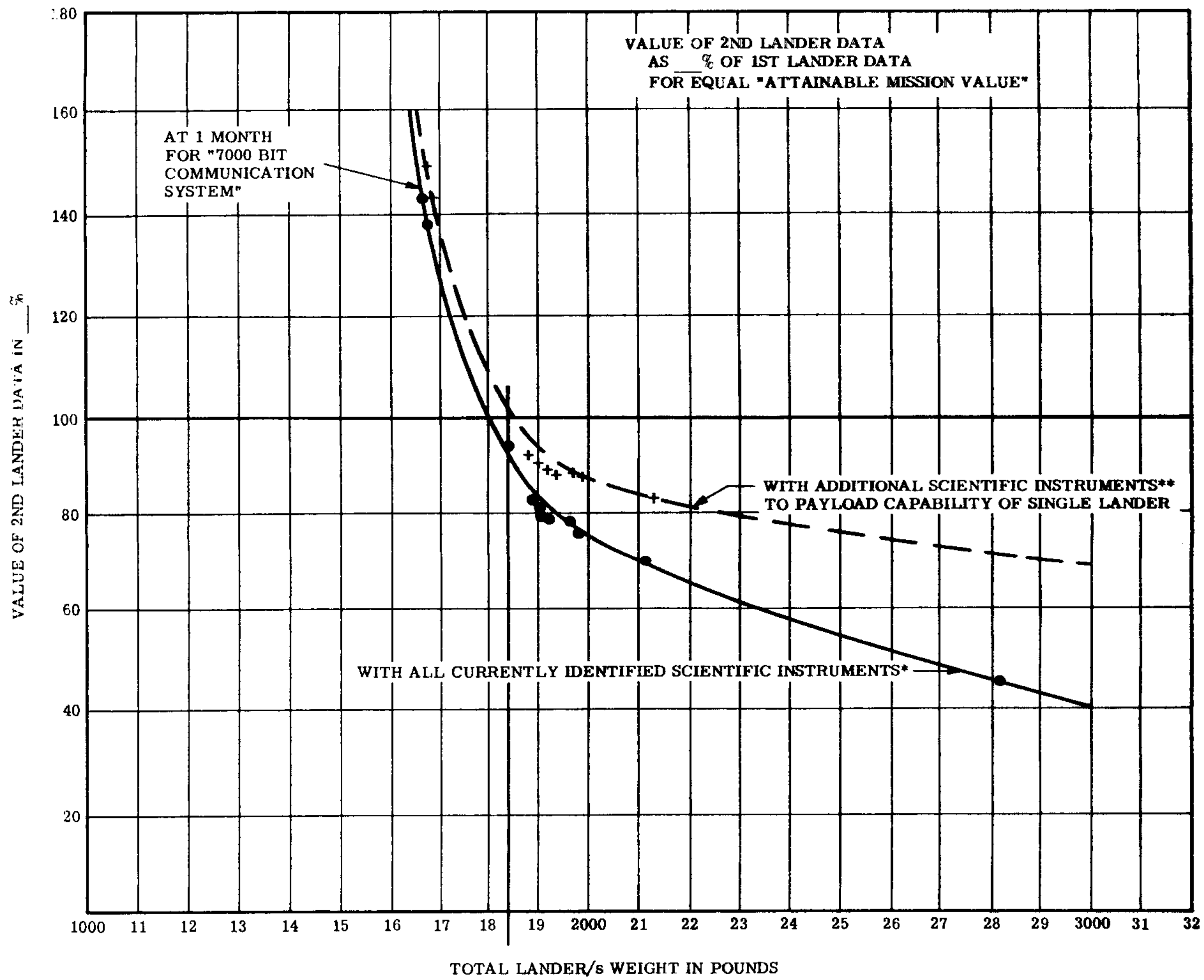


Figure 4.2.1-3 Attainable Mission Value at 24 Hours



*AS A RESULT OF REVIEWING THE TYPES OF INSTRUMENTS APPLICABLE TO THE MARS '69 MISSION--- NO ADDITIONAL SCIENTIFIC INSTRUMENTS WERE IDENTIFIED. THUS, THE TRADE-OFF CURVE ABOVE IS BASED UPON NO MORE INSTRUMENTS BEING ADDED TO THE SINGLE LANDER BEYOND THOSE ALREADY IDENTIFIED.

**IF ADDITIONAL INSTRUMENTS WERE TO BE IDENTIFIED FOR USE ON A LARGER SINGLE LANDER, IT IS ASSUMED THAT THEY WOULD CONTRIBUTE "MISSION VALUE PER POUND OF INSTRUMENT" AT A RATE EQUAL TO OR LESS THAN THAT OF THE LOWEST RANKED INSTRUMENT INCLUDED ABOVE --- NAMELY --- .03% PER LB. THIS FINE LINE IS CARRIED TO SHOW THE TRADE-OFF APPLICABLE FOR THIS .03%/LB. RATE.

Figure 4.2.1-4 Single vs Dual Lander

(7) Communications

The communications system includes the major portion of the electronic components of the system. Its complexity and the many functions it performs which are vital to the proper functioning of all the other subsystems make its reliability particularly important. Detailed analyses of the latest optimization completed during the study contract are detailed in later sections. However, it is of interest to note that while the communication rate capability is lower, the Lander reliability is greater using communication directly to the earth as compared to communicating via the Orbiter. This is illustrated in Figure 4. 2. 1-5.

(8) Reliability Requirements and Apportionments

The development of Markoff chain analyses to establish optimal numbers of launches, etc. is largely precluded by the serious restriction of launch opportunity. Thus, a cost effectiveness analysis is dominated by the major individual cost elements of the overall complex required per launch opportunity. These costs include those of the launch vehicle, the launch complex, the deep space information system logistics, and other elements in addition to those made available in the course of this study. From the initial analyses made under this study, it was concluded that the Voyager Vehicle System should be required to demonstrate a reliability and confidence prior to launch which would assure that mission success would be achieved in at least an average of 3 out of 4 launches as noted in paragraph 4. 2. 1A(2) above. This corresponds to a reliability of 65 percent, based upon exponential tables and methods of analysis (e. g. , T. I. S. R62SD135) with a lower limit confidence of 50 percent. As illustrated by Figure 4. 2. 1-6, the system proposed with two Landers is capable of meeting this requirement. See section 4. 5 for further detail.

In the columns of the Reliability Management Matrix provided in the Reliability Program Plan of Volume VI, a detailed reliability estimate to subsystem and component level is provided. This is also provided as section 6. 2 of S-31100 "Reliability Requirements for Contractors and Subcontractors." Since these calculated values are based upon the exclusive use of high reliability parts, materials, processes, etc. , which have been qualified and controlled by the best known techniques, it is felt that they are representative of the best demonstrable levels. Subject to the availability of cost information (noted above) and the completion of cost effectiveness analyses inclusive of them, the reliability estimates established by this study are considered applicable as minimum reliability requirements for the preliminary design and later phases of the Voyager Program.

B. Mission Effects of Other Reliability Factors

The success of the entire Voyager program is dependent upon many elements not included within the scope of the Voyager Spacecraft System for which the detailed reliability analysis and study has been prepared and documented in this report. A very significant, and perhaps the dominant factor of such elements, is the performance capability and reliability of the launch vehicles themselves.

(1) The probability of success for any launching for a Voyager is the product of the reliabilities of the Voyager Vehicle System as covered by this report and summarized in Figure 4. 2. 1-6 (above) and those of the launch vehicle, etc. of which the data in Figure 4. 2. 1-7 is considered to be representative.

(a) If the 81 percent (or greater) successful launchings, per Figure 4. 2. 1-7 are to be considered applicable to each the 1969-Mars opportunities of the Voyager missions, a period of operational testing of the components of the 1969 launch vehicle comparable to that which has been true of the components used in the systems from which Figure 4. 1. 1-7 was plotted must be provided. Without such opportunity to assure

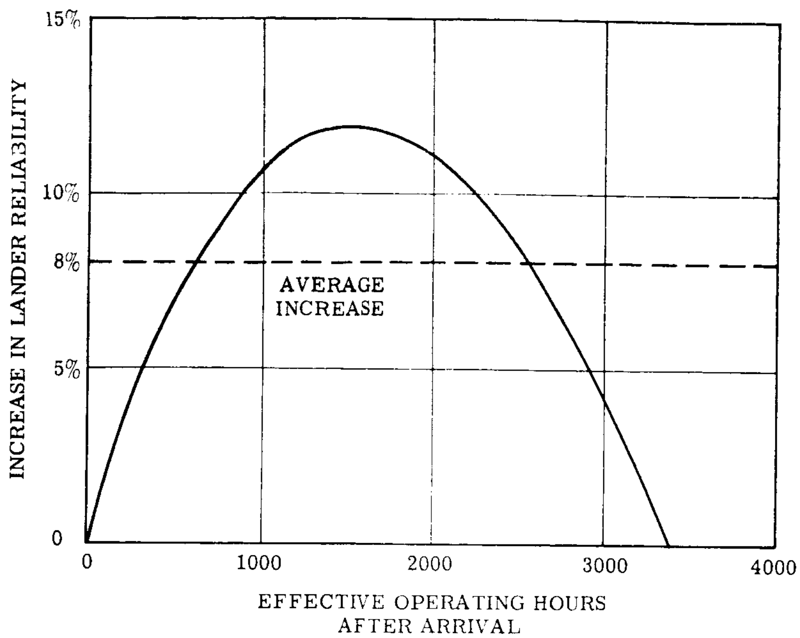


Figure 4.2.1-5 Percent Increase in Lander Reliability Using Communication Via Direct Link vs Via Orbiter To Earth

	10 Hours	100 Hours	1000 Hours	3 Months
Orbiter with one Lander	64%	62%	51%	34%
Orbiter with two Landers (one surviving)	71%	69%	57%	39%

NOTE: These are applicable to " continuous operation" after arrival. Other operational modes are detailed in the report.

Figure 4.2.1-6. System Reliability (100% Duty Cycle)

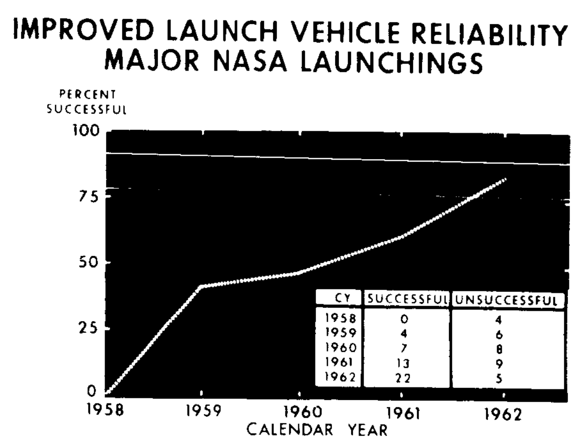


Figure 4.2.1-7 Improved Launch Vehicle Reliability - Major NASA Launchings.

the probability of launch success, the probability of mission success of the Voyager System cannot be verified.

(b) It is equally important that a comparable opportunity to demonstrate the performance capabilities and reliability of the Voyager Vehicle System be provided prior to launch if mission success is to be assured.

(2) Launch Opportunities

The launch opportunities periods are of limited duration (e. g., 30 days) and occur at widely separated intervals (e. g., 2 years). Thus, any lack of readiness to launch upon demand whether caused by malfunctions during count down due to the unavailability of reliable components either in the booster or the vehicle systems or by the operationsl procedures or personnel involved in the launch operation could well consume major portions if not all of any given opportunity period.

The two year slippage of the program opportunity which would result from such an interruption or delay could be very costly, including as it would not only the cost elements included in the Voyager Spacecraft Cost, but also large elements of the costs of booster system, RTG and scientific payload, etc. involved in the launch preparations. It is expected that the administrative and financial significance of such slippage costs make mandatory the launch of the best Voyager System operable provided that such criteria as immediate performance and sterilization are satisfied. It is essential that the program plan and schedule assure that such does not compromise the reliability of the systems launched.

Time is the most critical factor in the reliabilitiy area. Since the demonstration and verification of the reliability of the system design can only be begun when representative development hardware is available for that purpose, it is completely subject to any adverse variation in the almost numberless details of the design and development program.

From the reliability viewpoint, the most effective action which can be undertaken to increase the assurance of success and minimize program costs and risks is to advance the rate at which definitive design and development work is undertaken together with the associated materials and other reliability investigations and evaluation tests and verification which this makes possible. In effect, this is illustrated in Figure 4. 2. 1-8 by the transfer of as much of these activities as possible from period B to period A.

C. General

The methods and data as presented in section 4. 5. 1A(7) are directly applicable to the evaluation of the Titan IIIC system opportunities.

It is of importance to note that with the exception of the Orbiter and Lander combination with its relatively small payload capabilities, the reliabilities of two launches (namely, that two separately launched systems must both operate successfully) must be considered in making any simple direct comparison with the Saturn CIB System. However, simple, direct comparisons may be misleading because of the different payload capabilities are involved. Mission value comparisons are provided in the following paragraphs.

(1) Titan IIIC - Orbiter-Lander System

As noted in Table 3.10.2-1, the payload or net scientific instrument weight capability of single, small Lander is 80 lbs. As indicated in column 8 of Table 4.5.1-4a, this 80 lbs. of net scientific payload weight would permit the Lander to carry scientific instruments of priority No. 1 thru No. 23.

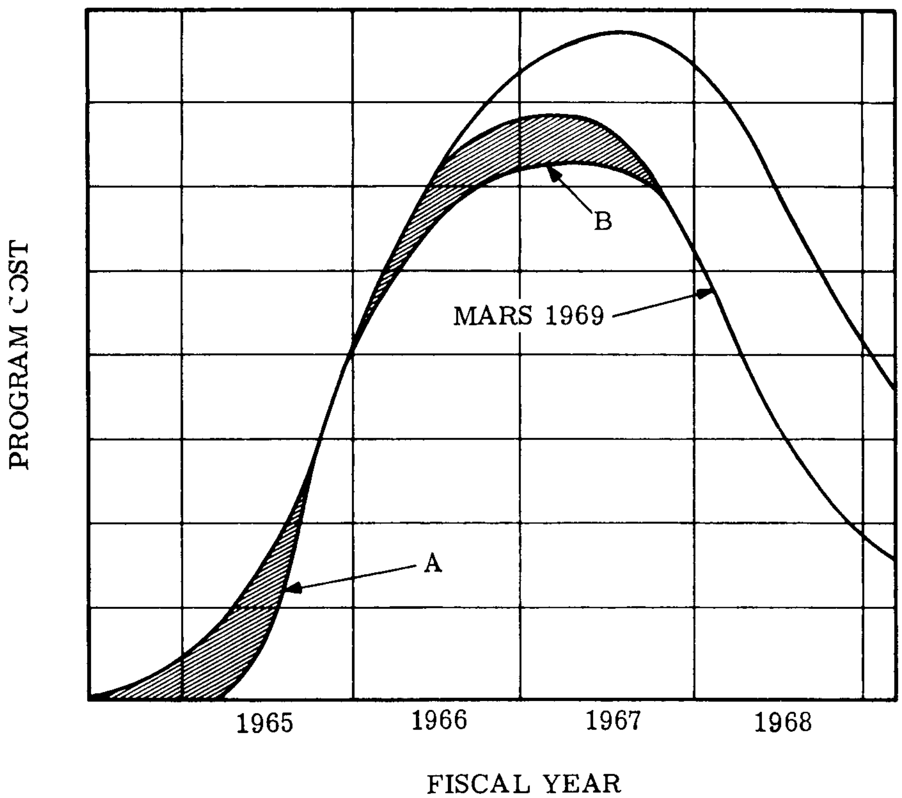


Figure 4.2.1-8 Voyager Spacecraft Summary

Per Table 4.5.1-3b, Lander instruments No. 1 thru No. 23 represent a 50 percent available mission value (Mars 1969) as compared to the Saturn CIB system at 60 percent for planet surface measurements as well as 8 percent as compared to 10 percent for the Saturn CIB for entry measurements.

The orbiter payload capability noted in Table 3.10.2-1 is 66 pounds. This would permit it to include per Table 4.5.1-3a, orbiter instruments of priority No. 1, 2, 3, 5, 6, 7, 8 and one TV camera together representing a 15 percent available mission value (Mars 1969) as compared to the 30 percent for the Saturn CIB Orbiter.

Thus a total available mission value of 73 percent of that available from a single lander and orbiter combination of the Saturn CIB would be available with the Titan IIIC Orbiter-Lander System.

Thus, while at first glance the small payload capabilities of this Titan IIIC Orbiter-Lander combination may seem to provide serious limitations, very significant mission values are available.

(a) Reliability Effects

The reliability of this Titan IIIC Orbiter and Lander system requires further definition of the equipment details before a mathematical model and analysis can be made comparable to that prepared for the Saturn CIB in section 4. However, since it must perform all the same maneuvers in transit etc. and has a somewhat simplified communications subsystem, its reliabilities are considered at this writing as equal to those of the corresponding elements of the Voyager system of the Saturn CIB.

By including these reliability considerations the corresponding available mission values per booster can be taken directly from Table 4.5.1-4.

These figures should, of course, be modified by including the launch reliability of the boosters and launch complexes as a function of the particular opportunity.

(b) Launch Reliability as a Function of Launch Capability:

The limited duration of the launch opportunity (e.g., 30 days) may result in as little as a single Saturn CIB launch being made per opportunity rather than a number of launches. With the long time interval (e.g., 20 days) expected to be required to make ready and launch a second Saturn Booster and Voyager System after firing a prior such launch while employing a two pad, single gantry launch facility and with the additional cleanup requirements (e.g., 10 days), the possibility of a third launch becomes quite doubtful. Thus, two Saturn CIB launches per planetary opportunity seems likely.

If the booster reliability per launch is 80 percent per per Figure 4.2.1-7, then the probability of obtaining one successful launch of a Saturn CIB from two launchings would be 96.0 percent.

Since it is to be expected that by 1967 or 1969 there will have been more opportunity to refine and improve the operational reliability of Titan IIIC than for the Saturn (including a required new 3rd stage for Voyager application), the Titan booster reliability should be equal or better than that of the Saturn Booster. However, for this writing they are considered to be the same.

The relatively short time interval (e.g., 12 days) expected to be required to make ready and launch a second Titan IIIC booster and Voyager system from a single pad (including cleanup time) and the availability of three launch pads make the possibility of a launch every four days a reasonable expectation. It is of interest then to consider that the Launch Reliability for one successful launch in a 30 day opportunity interval exceeds 99.9 percent.

(c) Comparison of Attainable Mission Values:

Since in a 30 day "window" there will be a capability for seven launches using the Titan IIIC boosters vs. two for the Saturn CIB, the data of Table 4.2.1-1 provides a basis for considering may alternatives. One such alternative is to consider the mission value attainable by the launching of seven Titan IIIC boosters each with a single Lander and Orbiter system aboard.

Considering each successful system of equal mission significance and using the reliability values for the Vehicle as shown in Figure 4.2.1-7 and combining this with the launch reliability per Titan (or per Saturn) provides an overall reliability per system of approximately 50 percent. Using this value to enter Table 4.2.1-1 and using the mission value of the small Orbiter and Lander payloads for the Titan IIIC launch as noted above, we have the following attainable mission values:

As indicated by Table 4.2.1-2, the attainable mission value by using four Titan IIIC "Orbiter with single Lander" systems begins to exceed that attainable using two Saturn CIB systems.

Since the success of any prior launch ing cannot be known prior to the last launch opportunity in any given year (because of the long transit time to the planet), any consideration that might be given to different combinations or evaluations of scientific instruments must be done upon the basis of "probable" successes from previously successful launches and midcourse maneuvers

(2) Titan IIIC Orbiter With Separately Launched Titan IIIC Dual Landers

Table 3.10.2-1 indicates that an Orbiter payload capability of 223 lbs. is available for an Orbiter (with no Landers) launched by a Titan IIIC. This provides an Orbiter with complete communications and mapping capability. Its available mission value as shown in Table 4.5.1-3a is 30 percent. While this is much lower than the 73 percent available value per Titan noted in 4.2.1.C(1), it allows the subsequent Titan IIIC's to be used for Landers only.

As mentioned in Section 3.10 there is a slight gain in reliability in each of these systems as compared with that of a Saturn Orbiter and a single lander system because of the simplification of transit maneuver requirements. However, from a review of the mathematical model and reliability analysis in section 4.5.2, it is considered that this should at present be neglected and the same reliability should be used as is applicable to the Voyager systems planned for the SATURN CIB.

Table 3.10.2-1 indicates a gross Lander weight of 2230 lbs. in the Bus/Lander system for Titan IIIC. Since this is considerably greater than the 1840 pounds point in Figure 4.2.1-3 above which it is of clear advantage to apply dual landers, the payload capability of each of two landers having a gross weight of 1115 lbs. has been determined as 85 lbs. This is sufficient to include instrument #24.

For such a dual lander combination and using the same reliability values for the Titan as for the Saturn dual lander system, as noted above, the Attainable Mission Value may be taken directly from column 29 of Table 4.5.1-4. Including instruments of priority #1 to 24, this is 63.5%. Correspondingly, from column 34, the Attainable Mission Value from the separately launched Orbiter is 19.8%.

With the first successful Titan launch devoted to the Orbiter, Table 4.2.1-1 may be applied to subsequent launches of dual lander vehicles. (It should be noted that the window for "lander vehicles" extends well beyond the 30 days available for launching "orbiter vehicles".) Thus, we have the following Attainable Mission Values:

TABLE 4.2.1-1

PROBABLE SUCCESSFUL MISSIONS / LAUNCH ATTEMPTS	ORBITER & SINGLE LANDER TITAN IIIC	ORBITER & SINGLE LANDER SATURN CIB	ORBITER & DUAL LANDERS SATURN CIB
1/1	44%	63%	78.8%
1/2	66	94.8%	118.2%
2/2	44	63%	78.8%
1/3	77	No opportunity available	
2/3	88.1	" "	"
3/3	33.3	" "	"
1/4	82	" "	"
2/4	121	" "	"
3/4	82.5	" "	"
1/5	85	" "	"
2/5	143	" "	"
3/5	132	" "	"
2/6	157	" "	"
3/6	173	" "	"
4/6	120	" "	"
2/7	165	" "	"
3/7	204	" "	"
4/7	176	" "	"
3/8	226	" "	"
4/8	224	" "	"
5/8	160	" "	"

TABLE 4.2.1-2. PROBABILITY OF SUCCESS OF AT LEAST "S" VOYAGER SYSTEMS FROM A NUMBER OF "n" LAUNCHINGS OF BOOSTER + VEHICLE RELIABILITY "R"

n	s	R = 90%	80%	70%	60%	50%
2	1	.990	.960	.910	.840	.750
	2	.810	.640	.490	.360	.250
3	1	.999	.992	.973	.936	.875
	2	.972	.896	.784	.648	.500
	3	.729	.512	.343	.216	.125
4	1	.999	.998	.991	.974	.937
	2	.996	.972	.916	.820	.687
	3	.947	.819	.651	.475	.312
	4	.656	.409	.240	.129	062
5	1	.999	.996	.997	.989	.968
	2	.999	.993	.969	.912	.812
	3	.991	.942	.836	.682	500
	4	.918	.737	.528	.336	.187
	5	.590	.327	.168	.077	.031
6	1	.999	.999	.999	.995	.984
	2	.999	.998	.989	.959	.890
	3	.998	.983	.929	.820	.656
	4	.984	.901	.744	.544	.343
	5	.885	.655	.420	.233	.109
	6	.531	.262	.117	.046	.015
7	1	1.0	1.0	.999	.998	.992
	2	1.0	1.0	.996	.981	.937
	3	.999	.995	.971	.904	.773
	4	.997	.966	.874	.710	.500
	5	.974	.852	.647	.419	.226
	6	.850	.577	.329	.159	.062
	7	.478	.210	.082	.028	.008

From Table 4.2.1-3 and adding one orbiter Titan IIIC booster to those applicable to Titan IIIC in the table, it is evident that at four (4) Titan IIIC boosters a crossover point is obtained as in Table 4.2.1-2.

(3) Comparison of Titan & Saturn Systems

Figure 4.2.1-11 presents graphically the data in Tables 4.2.1-2 and -3.

Other combinations of scientific instruments in given orbiters or landers can be made to obtain specific information but which will lower the Titan IIIC curve somewhat from its maximum position shown in Figure 4.2.1-9. Many alternatives with notable flexibility and adaptability are available. Upon a cost basis as well as on an Attainable Mission Value basis, there appears to be very significant advantages to the use of the Titan IIIC. It also would be of significance to re-examine launch opportunities earlier than 1969 for Voyager application including the new Martian atmospheres and requirements.

4.2.2 CONCLUSIONS

During the Study Contract, the Reliability & Quality Assurance requirements for the Voyager System and Program have been considered in detail and both the technological data, methods and approaches, and the organizational and management alternatives and procedures necessary for the successful attainment of the Voyager Program's objectives have been examined and evaluated. These analyses and their principal recommendations and conclusions as they have been developed have been presented to the Voyager Program managers, scientists, and engineers at frequent intervals. These conclusions and recommendations included:

1. That the reliability requirements for the Voyager System be made quantitative and that the capability of each component and subsystem to fulfill these requirements be required to be demonstrated by tests prior to the scheduled launch of a Voyager System.

2. That these reliability requirements be so defined including both "reliability" (see paragraph 4.2.3, item 1) and "confidence" (see paragraph 4.2.3, item 2) as to assure that an average of three (3) successes out of four (4) opportunities may be expected to be attained by the "Voyager System" (see paragraph 4.2.3, item 3).

3. That the demonstration of these reliability capabilities over the extensive time periods involved in the interplanetary flight and planetary operational times be required to be provided only at basic part, material, and process levels and that this basic level of demonstration be so integrated by the prime contractor as to avoid duplication of expenditures by subcontractors (or the prime contractor) and so as to assure the rapid availability of test results to all Voyager contractors having need for this information. Also, that this integration of effort include the investigation of and making available to each Voyager contractor of such information as may be applicable to Voyager from other current and prior space and defense programs.

4. That the demonstration of these reliability capabilities for all ranges of variation and extremes of environment and loading which may be anticipated during an actual mission (whether occasioned by factors external to or internal to the Voyager System) be required by tests conducted at "system level" (see paragraph 4.2.3, item 4) for sufficient periods of test time to assure complete response of the Voyager System components to these loadings. That these system level tests contain a sufficient number of "cycles" (simulating the magnitudes and sequence of such loadings as

TABLE 4.2.1-3.

PROBABLE MISSIONS SUCCESSFUL/ LAUNCH* ATTEMPTS	SEPARATE ORBITER PLUS DUAL LANDER LAUNCHINGS TITAN IIIC		ORBITER & DUAL LANDER SATURN CIB
1/1	55	16.2 = 69.2	78.8%
1/2	79.5	16.2 = 95.7	118.2
2/2	53	16.2 = 69.2	78.8
1/3	92.8	16.2 = 109	No Opportunity Available
2/3	106	16.2 = 122	" " "
3/3	40	16.2 = 56	" " "
1/4	99	16.2 = 115	" " "
2/4	146	16.2 = 162	" " "
3/4	99.6	16.2 = 116	" " "
1/5	93.2	16.2 = 109	
2/5	172	16.2 = 188	
3/5	159	16.2 = 175	
4/5	80	16.2 = 96	
2/6	189	16.2 = 205	
3/6	20	16.2 = 225	
4/6	146	16.2 = 162	
2/7	199	16.2 = 215	
3/7	246	16.2 = 262	
4/7	212	16.2 = 228	

*With a launch reliability of 80% or greater and four (4) days between launchings, the probability of successfully launching one out of two orbiters is very high. The Attainable Mission Value of 16.2% for a single launch attempt has been included in the TITAN IIIC Column. In making cost comparisons, the cost of these orbiter launchings must be added to the launch attempts in Column #1.

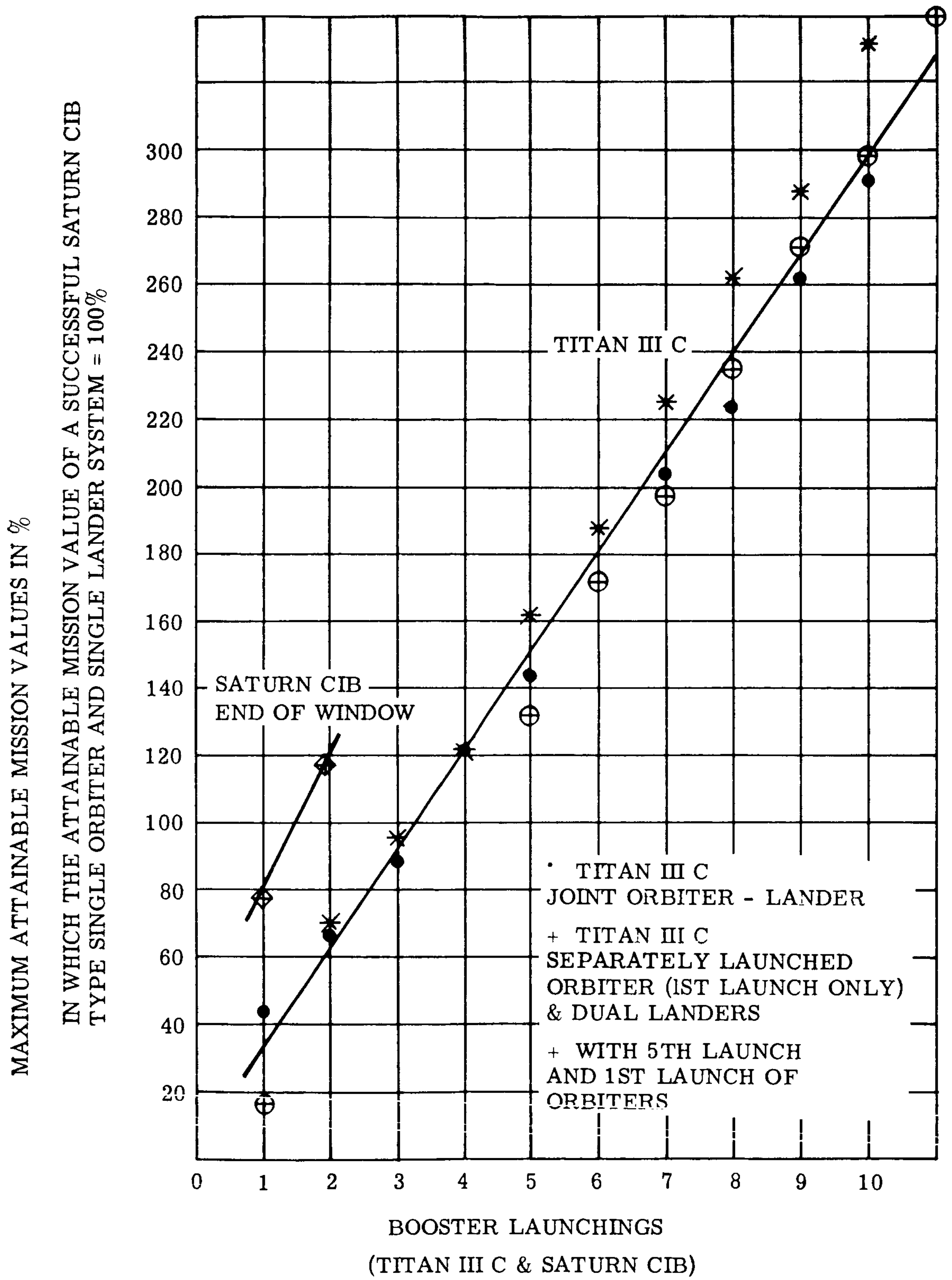


Figure 4.2.1-9 Attainable Mission Values for Multiple Launchings of Satırn vs Titan Booster Voyager Systems

may be expected to occur during an actual mission including all effective factors from the manufacturing and test periods and from the interplanetary and planetary operational periods) to demonstrate the specified reliability of each component of the system to its required level of confidence.

5. That the suitability of each component to enter a systems test be required to be demonstrated prior to its incorporation into any system intended for qualification, acceptance, or reliability demonstration tests. That this demonstration include satisfactory operation under thermal-vacuum conditions (per item 4 above) for not less than 150 hours of active operational testing, of which the last 100 hours are required to be FAILURE FREE as a condition for the satisfactory conclusion of the demonstration.

   And that the suitability of each system be required to be demonstrated by test prior to its acceptance for shipment and use. That this demonstration include the satisfactory operation of each component of the system under thermal-vacuum conditions (per item 4 above) for not less than 1000 hours, of which the last 700 hours are required to be FAILURE FREE as a condition for the satisfactory conclusion of the demonstration.

   Test program requirements incorporating these provisions are provided in section 4 of Specification S-31100 "Reliability Requirements for Voyager Contractors and Subcontractors" which has been prepared under the study contract and of which additional copies are available upon request.

6. That the final design of the Voyager System be required to be optimized, as it has been during this study contract, to provide a "maximum system effectiveness" (see paragraph 4. 2. 3, item 5) per launch. The approach and formulae used to assure its quantitative definition and attainment are provided later in the report. And, that these cost effectiveness analyses which have been (or will later be) made be so prepared as to include all significant program costs elements (including those not directly included in the scope of the "Voyager System" (see paragraph 4. 2. 3, item 3 of this study contract) and provide an evaluation of cost effectiveness in terms of "maximum system effectiveness per dollar."

7. That the apportionment of reliability requirements to each Voyager System component and subsystem be based upon detailed reliability analyses of all significant system elements in which the effects upon system reliability of the design margins "worst case" design limits, and safety factors used in the design, as well as of each practicable, alternative mode of operation and "back-up" (see paragraph 4. 2. 3, item 6) component as they provide effective redundancy in increasing the reliability of the Voyager System, have been considered.

   Thus, additional redundancy, simplification or other research, design and development or reliability engineering efforts are to be applied to those portions of the system which provide the greatest opportunities for increased systems reliability per pound of systems weight required for their implementation. Such an apportionment to sub-subsystem/component level has been completed during the study contract and is appended as section 6. 2 of Specification S-31100 "Reliability Requirements for Voyager Contractors and Sub-contractors."

8. That, in order to assure the attainment of maximum system effectiveness this apportionment be optimized, during preliminary and final design as it has been during this study contract, to attain maximum reliability per pound of weight of the system.

9. That sets of "Approved Parts and Materials Lists" and "Design Standards" be established for the Voyager Program through the participation of each principal subcontractor's engineers with those of the prime contractor and with NASA, and that conformance of the Voyager System design to the Approved Parts, Materials and Standards be incorporated as requirements under the contract. This is recommended in relation to item 3 and item 6 above to assure that the best parts and materials, and the processes and methods, selected for their application to the component and system designs, are consistently used by all component and system design and development engineers for each component to which they are applicable.

   An initial issue of such an approved parts list is appended as section 6. 3 of Specification S-31100.

10. That each contractor and subcontractor be required to establish suitable facilities and organization including responsibility and authority definition and delegation to assure the accomplishment of the reliability requirements of the Voyager Program. That this be documented as a portion of that contractor's Reliability Program Plan and that such a plan be prepared and implemented in accordance with NASA document NPC 250-1.

    Such a requirement is provided by section 3. 1 of Specification S-31100.

11. That for the Mars 1969 opportunity, two (2) Landers be used in each launch in which a complete "Voyager System" (see paragraph 4. 2. 3, item 3) is involved and that they be so provided with guidance and control as to assure their arrival on two significantly separate planet locations each of which will provide information unique to that location.

12. That for the Mars 1969 opportunity direct link communication to earth be provided on each Lander in addition to the communications provided via the Orbiter.

13. That cost effectiveness and reliability data applicable to the Saturn boosters, launch facilities, DSIF, logistics, etc., and from the results of this study contract (s) be made available and considered together with a more detailed study of the possible applicability of the Titan III booster, launch facilities, etc. to the Voyager Mars 1969 and other opportunities in an overall cost effectiveness study prior to or as a part of the preliminary design phase of the Voyager program.

14. That the Voyager Lander Parts Sterilization Compatibility Program be initiated as a part of the preliminary design phase of the Voyager program.

4. 2. 3 DEFINITIONS

1. Reliability – the probability of successful operation of the Voyager System from time of launch through a given point in time during a Voyager mission.

2. Confidence – the probability (expressed in percent obtained from prior testing and performance demonstration) that a like unit will have a reliability equal to or greater than that stated.

3. Voyager System – the entire interplanetary vehicle including Orbiter and Lander(s) but not including the Saturn Boosters, DSIF, or other ancillary systems.

4. "System Level" Tests - include not only the test results obtained when a complete system is operationally tested but also those test results including qualification and prototype tests in which system interfaces, loadings, sequences or cycles, and transfer functions are so well simulated as to provide equally valuable data for establishing the performance and reliability capabilities of these systems components.

5. The term "maximum system effectiveness" as used above is to be interpreted as assuring a "maximum probability of obtaining and communicating to earth the scientific data from the instruments carried in such a combination as to provide maximum scientific value relative to that Voyager mission's specific flight objectives."

6. The "back-up" term is applicable to any component or mode of operation which may be called upon to provide satisfactory performance in the event that the principal or primary unit or mode does not provide suitable performance.

4.3 SIGNIFICANT RELIABILITY FACTORS

Many features designed into the system make significant contributions to the probability of maintaining successful operation of the system throughout the long mission period. Some of the outstanding features are described in the following sections.

4.3.1 COMMUNICATIONS

The communications subsystem is designed with interlocking links and backup modes of operation to the extent that a minimum of three failures must occur in the Orbiter-dual Lander-Earth link in order to completely fail the function in the orbit-Lander phase.

Figure 4.3.1-1 shows the proposed linkage which is used to illustrate the tolerable failure combinations which can occur without loss of data transmission.

The Hi-Gain Antenna System is used as a backup to the Omni System during the first part of the transit phase. Then the Omni System is used as a backup for the Hi-Gain Antenna System during the remainder of the transit phase and during the entire orbit phase.

Two receivers in the VHF System of the Orbiter are in operation only during the separation-to-impact phase of the Landers.

The thermo plastic recorders are redundant except when high rates of data acquisition are required.

During the orbit phase, communications is on approximately 2 hours per day for the Orbiter-Lander link.

4.3.2. POWER SUPPLY-ORBITER

Several significant factors in the power supply design contribute to the inherent high reliability of this subsystem.

The solar array assembly requires thousands of individual solar cells which are assembled in a matrix selected to provide the most efficient failure free arrangement of cells. The solar array contains a surplus of cells to allow for possible loss of power due to the degradation of the array in a given time period through radiation damage, variation of solar constant, micrometeorite damage, manufacturing and filter losses, and random cell failures.

Protective diodes are used in the solar array to isolate the effect of cell failure within any one submodule in a series string, and redundant parallel diodes are used to protect the system in the event of a shorting failure across the series-connected modules.

The battery in the power supply subsystem is utilized only during peak power periods or during the occurrence of any dark period.

4.3.3 GUIDANCE AND CONTROL-ORBITER

This subsystem also has some significant reliability factors, most of which involve backup modes of operation.

The failure of the narrow Sun Sensor degrades orientation but the function is still possible with less accuracy with the primary sun sensors.

The earth tracker, via the programmer, can be used as backup for the Canopus tracker and vice versa.

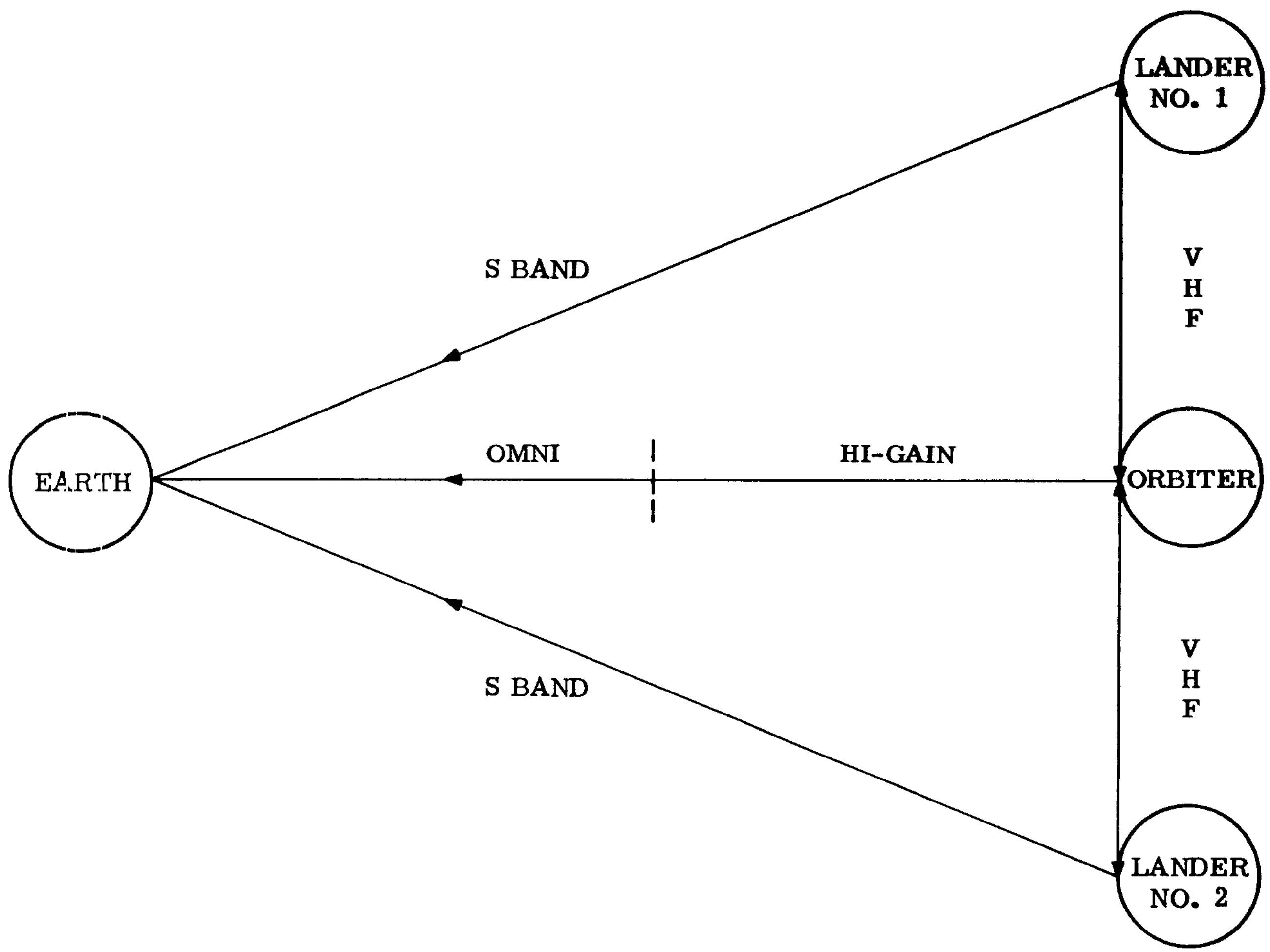


Figure 4.3.1-1 Orbiter - Dual Lander - Earth Communications Link

If the planet sensor fails, PHP can be oriented by pre-programmed data.

All axis amplifiers are in total redundancy.

An override command capability exists for programmer functions.

4.3.4 STRUCTURE AND HEAT SHIELD-LANDERS

GE/MSD structural and heat shield design practices provide reliability through the use of large safety factors and low strength variance of materials. Structural design adequacy and heat shield integrity will be assured through a thorough test program to demonstrate stress strength and fatigue strength safety margins as well as heat shield and bond strength and back-face temperature characteristics.

4.3.5 INSTRUMENTATION AND SCIENTIFIC EXPERIMENTS

All instrumentation and the experiments will be incorporated into the vehicle design such that loss of any one experiment will not cause complete loss of all the scientific data, nor cause secondary failures to occur. GE/MSD is prepared to assist in the evaluation of the design reliability of the experiments in order to assure maximum return of scientific information.

4.4 CRITICAL PROBLEM AREAS

During the course of this study, several critical problem areas emerged which require further study and/or development programs. Table 4.4-1 lists some of these problem areas and also possible methods of obtaining solutions to these problems.

TABLE 4.4-1. CRITICAL PROBLEM AREAS

PROBLEM	EXAMPLE	SOLUTION
Lifetime capability of new type components presently in the development stage	TP Recorders Image Orthicons Klystrons Hi-Gain Antennas Star Trackers	Development Program and Life Tests
Complexity of parts in selective components resulting in a low survival probability	Star Trackers Power Amplifiers Data Processor Programmers	Majority Logic Redundancy or total component redundancy
Effects of Sterilization on sensitive parts or components	Batteries Vidicons Explosive Devices Propellants	Study and Development
Environmental conditions which may prevail in long term missions or planetary atmosphere	Solar Flares Radiation Belts	Study and results of space program prior to Voyager

4.4.1 DESIGN IMPROVEMENTS

Design deficiencies and critical problem areas which would seriously influence the required performance were investigated and corrected during the many design iterations. These design improvements greatly increased the inherent reliability of the proposed system. Some of these design improvements are listed in Table 4.4.1-1 with an indication of the increase in component reliability.

TABLE 4.4.1-1. DESIGN IMPROVEMENTS

Component	Design Improvement	Reliability Increase	
		From	To
Star Tracker	Simplification	.84	.98
Power Amplifiers	Simplification and Component Redundancy	.92	.99
Storage & Logic Unit	Majority Logic Redundancy	.82	.97
Processor	Majority Logic Redundancy	.82	.99
Command and Computer Equipment	Majority Logic Redundancy	.94	.98
S-Band Transmission Loops (Orbiter to Earth)	Simplification and Component Redundancy	.94	.99

4.4.2 ENVIRONMENT

A most pressing problem existing in the mission is the radiation intensity which may be encountered by the occurrence of solar flares or unknown radiation belts in the approach to Mars. If the flare flux level is intense or an extended time period is spent in an unknown intense planetary radiation belt, the effect on system operation could be abortive, if adequate safeguards are not provided.

Recognition of this problem by MSD in past space programs has led to in-plant radiation testing and other source data on parts and assemblies to investigate:

1. Part threshold of damage levels
2. Effect of low level flux for extended time periods
3. Combined radiation/thermal-vacuum data
4. Stable part parameters under radiation
5. Analytical prediction techniques
6. End of life parameters of parts

The usual treatment of stresses placed on parts or equipments by a known environment is to reduce it by shielding, dissipate it from the device, or design around some stable part parameter. All of these protective measures are based on knowledge of the approximate range of the environment the equipment will see, as well as the failure mechanism induced in parts by the environment.

However, the various effects of combined Radiation/thermal-vacuum environment are just beginning to be investigated. Therefore, present day design techniques cannot be assured as providing an adequate safeguard to equipment until more knowledge is acquired in this area. It is anticipated that by the time final hardware definition is contemplated for the first Voyager flight, these stresses will be more fully understood and processes developed to overcome their effects or reduce them to such levels as are fully consistent with the Voyager System's reliability requirements.

4.4.3 COMMUNICATIONS

The reliability of the communications subsystem design is complicated by problem areas which require special attention and investigation. Some of these problems are listed in Table 4.4.3-1 with suggested means of solution.

TABLE 4.4.3-1. COMMUNICATION PROBLEM AREAS

Problem Areas	Means of Solution
1. Unknown behavior of thin film devices and thermoplastic recorder in space environment.	1. Reliability techniques such as parts screening, derated parts application, and protection against adverse environments, followed by an adequate test program to demonstrate capability prior to flight.
2. Long life limitations of parts in space.	2. Paragraph (1) above plus low percent duty cycle.
3. Stresses imposed by cycling	3. Keep cycling rate down to a minimum, allow relatively long periods of non-operation.
4. Complexity of design	4. Majority logic design techniques, and worst casing and isolation against failure due to most probable failure modes. Minimize effect of isolated failures on system effectiveness.

4.4.4 PROPULSION AND SEPARATION

Problem areas and means of solution are presented in Table 4.4.4-1.

TABLE 4.4.4-1. PROPULSION AND SEPARATION

Problem Areas	Means of Solution
1. Leakage of cold gas system and hot gas system, prior to actual usage.	1. Proper design against leakage plus an adequate demonstration test program.
2. Corrosion of components in hot gas system due to on-off cycling during long transit period.	2. Proper design against corrosion, plus an adequate demonstration test program.
3. Long term storage of pyrotechnics in vacuum and radiation of space.	3. Proper design, selection of parts plus protection against severe environment.
4. Assure performance when required.	4. Redundant squibs and explosive bolts, IFD, and squib valves. Pyrotechnic lot screening plus demonstration test program.

4.4.5 RETARDATION

Some problem areas in this subsystem and suggested means of solution are presented in Table 4.4.5-1.

TABLE 4.4.5-1. RETARDATION SUBSYSTEM

Problem Areas	Means of Solution
1. Parachute materials problem — due to high velocity and temperature.	1. Extensive design and test program to develop adequate parachute system.
2. Environment uncertainty and Mars atmosphere.	2. Design for wide range of conditions. Have backup sensing and initiation systems-radar altimeter, baro-switch.
3. Long term storage of mechanical, electromechanical, and pyrotechnics in space before required to operate.	3. Isolate as much as possible from adverse environment. Use redundant ignition of squibs and redundant squibs for IFD., explosive bolts, fittings, and mortar. Redundant reef line cutters is standard practice.

4.5 SYSTEM ANALYSIS

4.5.1 VOYAGER SPACECRAFT

The Voyager Spacecraft is required to have the capability for both orbiting and landing on either Mars or Venus during the time period of 1969-75.

Any number of missions and mission profile variables can be applied to these various requirements and capabilities, since the mission times are distributed according to selected launch dates. The extremes of the mission intervals are 246 days for Venus and 390 days for Mars. The worst case period of 390 days for the Mars mission is used in this reliability analysis to present an indication of the probability of success that can be expected with present day part and design technology.

To acquire the maximum scientific data from both the Martian space and surface environments, the design concept of an Orbiting Vehicle and two redundant Landers was selected after assiduous trade-off analyses.

The two major functional systems in the overall vehicle, Orbiter and dual Lander, were then designed to obtain maximum capability for acquisition and transmission of this data.

The subsystems of both the Orbiter and any one Lander are shown in Table 4.5.1-1.

TABLE 4.5.1-1. ORBITER AND LANDER SUBSYSTEMS

Orbiter	Lander (2)
1. Communication	1. Communication
2. Power Supply	2. Power Supply
3. Guidance and Control	3. Propulsion (orbit ejection)
4. Propulsion	4. Instrumentation
5. Instrumentation	5. Experiments
6. Experiments	6. Retardation
	7. Thermal Control
	8. Orientation

The phases and time intervals per phase used in this analysis to gain some measure of mission success are shown in Table 4.5.1-2.

The Orbiter is designed to acquire information about the space environment during transit and in its 3 month orbiting interval. It serves as the primary communications link to Earth since it contains a command link and a rapid bit rate transmission capability. It will therefore be used as the relay link between the dual Landers and Earth stations for rapid scientific data transfer.

In addition to the orbit-Earth rapid communications link, a slower bit rate transmission capability to Earth is contained in the S-band communication subsystem in each Lander. Orbiter-Lander communication is maintained by the VHF link. Thus, Orbiter-Lander-

TABLE 4.5.1-2. PHASE AND TIME INTERVALS

Phase	Time Interval
Launch	10 hours
In-Transit	290 Days (280 days + 10 day contingency)
Orbiting	100 Hours and 3 months
Planetary Entry	One day
Surface	6 months

Earth communication can be maintained by the Orbiter or any one of the Landers and only the failure of all three systems would abort data acquisition and transmission.

The redundant Landers will have the capability for six months of operation after impact, although the majority of the required scientific data can be obtained in a shorter time interval.

A. SYSTEM ANALYSIS

(1) The systems reliability analysis represents the periodic summation and interpretation of the many subsystem and component analyses including the effects of their individual operating times, environments and the effects of backup modes and redundancies incorporated in the system design as a result of the failure effects analyses. For the subsystems comprising a Lander system, the reliabilities have been summarized and presented in Figure 4.5.1-1. The upper edge of each curve represents the Lander system reliability without including "scientific instrument reliability." The width of each curve illustrates the effect of a 20 percent complexity allowance for scientific instrument reliability.

(2) In order to develop a basic reliability analysis which is adaptable to any particular mission, the individual analyses are prepared relative to mission phases. Two principal "cut-off" or evaluation points in the mission cycle are taken to be at 100 hours and at 3 months after the arrival of the Landers.

(3) Although the list of scientific instruments in Table 4.5.1-3 is considered to accurately represent the mission objective and implementation, the individual composition and construction of each instrument is not sufficiently well known to permit its reliability estimate to be prepared. However, the 20 percent complexity allowance for scientific instrument reliability is considered more than sufficient.

This is very evident when each instrument is compared in complexity with the Lander system and again when we recognize that each instrument is independent of the others and that, considering the backup and secondary modes of interpretation applicable to TV panorama and to TV microscope data, no single instrument mode represents more than 5 percent of the Mission Value. Thus the failure of a single instrument does not represent a mission failure.

Applying a criteria of 75 percent Scientific Data Value return as a definition of mission success would allow as much as 5 or more instrument mode failures before mission suc-

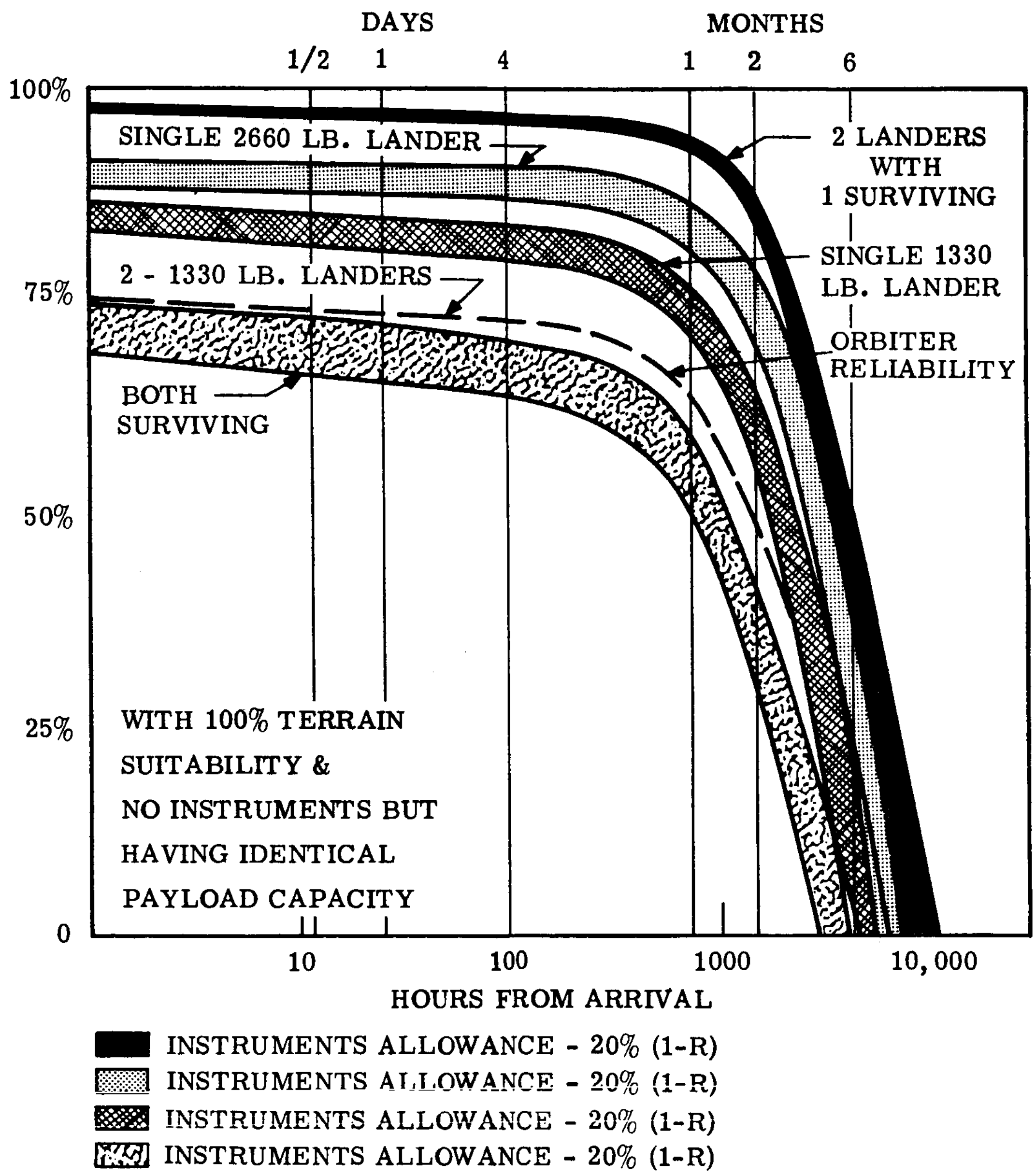


Figure 4.5.1-1 Lander Systems Reliability

TABLE 4.5.1-3a. SCIENTIFIC INSTRUMENTS

ORBITER Name of Instrument	Instrument Identifying Number	Total Orbiter Scientific Value Available (in percent)	ORBITER VALUE AVAILABLE VS TIME AFTER ARRIVAL (in percent) 12 Hours	24 Hours	96 Hours	30 Days	60 Days
1. Magnetic Field	I-23	2	75	85	95	98	99+
2. IR Flux	I-2	1	25	45	65	85	95
3. Radiameter UV to IR	I-79	1	75	85	95	98	99
4. TV (Multicolor)	TV	20	20	40	60	80	98
5. Charged Particle Flux	I-12	1	75	85	95	98	99+
6. Far UV — Radiameter	I-96	1	75	85	95	98	99
7. Micrometeoroids	I-55	1	60	75	90	98	99+
8. Ionospheric Profile	I-85	1.5	75	85	95	98	99
9. Polarimeter	I-95	.5	75	85	95	98	99
10. IR Spectrum	I-1	1	60	75	90	98	99+
Orbiter Subtotals		30					

TABLE 4.5.1-3b. SCIENTIFIC INSTRUMENTS

LANDER: Name of Instrument	Instrument Identifying Number	Total Entry Value Available (in percent)	Total Surface Value Available (in percent)	SURFACE VALUE AVAILABLE TIME AFTER ARRIVAL (in percent) 12 Hours	24 Hours	96 Hours	30 Days	60 Days
1. Temperature	I-24	1	3	50	60	90	95	98
2. Sounds	I-34	-	3	75	85	90	95	99
3. Pressure	I-17	1/2	1/2	90	92	97	99	99+
4. Density	I-20	2	4	50	60	90	95	98
5. Multiple Chamber	I-54	-	10	75	90	95	99	99+
6. Surface Penetration Hardness	I-25	-	2	95	95	95	98	99
7. Photoautotroph	I-62	-	3	75	90	95	99	99+
8. Light Intensity (Sun Sensor)	I-84	-	1/2	50	75	90	93	99
9. Composition, H_20	I-44	1/2	1/2	90	92	97	99	99+
10. Composition, 0_2	I-45	1/2	1/2	90	92	97	99	99+
11. Turpidity & PH	I-53	-	3	75	90	95	99	99+
12. Wind Speed & Direction	I-67	-	2	40	60	80	90	95
13. Gas Chromatograph	I-8	2	2	90	92	97	99	99+
14. Composition, N_2	I-48	1/4	1/4	90	92	97	99	99+
15. Composition, $C0_2$	I-49	1/4	1/4	90	92	97	99	99+
16. Soil Moisture	I-70	-	1	95	96	97	98	99
17. T.V. Camera, Panorama	T.V.	-	10	90	91	92	93	95
18. Radioisotope	I-19	-	3	75	90	95	99	99+
19. Composition, 0_3	I-46	1/4	1/4	90	92	97	99	99+
20. Composition, A	I-47	1/4	1/4	90	92	97	99	99+
21. Precipitation	I-36	-	1/2	25	50	60	80	90
22. Electron Density (Langmuir Probe)	I-39	1/2	-	-	-	-	-	-
23. Surface Gravity	I-72	-	1/2	99	99+	99+	99+	99+
24. Surface Roughness Altimeter (Pulse Radar)	I-5	2	-	75	85	90	95	99
25. Microscope, Including TV Camera, Drill, Handling Pulverizer, Sample	I-71	-	9-1/2	80	90	95	99	99+
26. Seismic Activity	I-21	-	1/2	50	70	80	95	99+
Lander Subtotals		10	60					

cess was seriously compromised. Thus "5 times" redundancy is applicable to the total composite of scientific instruments. Since the individual scientific instruments may be expected to have reliabilities (provided they survive their "storage" in vacuum during transit and 125 g's at impact) well above 90 percent out through the 100 hour point, the effective reliability of the composite to this point will be very high. The figures used in Tables 4. 5.1-4 for instrument reliability are therefore considered to be very conservative.

(4) System effectiveness may be measured in the same terms of Attainable Mission Value as discussed in Paragraph 4.2.1. A(5) Attainable Mission Value is the product (i.e., integrated) of the applicable system reliability and the increment of value, in percent, of the complement of scientific instruments as a portion of the total value available through the complete mission period.

By combining at given time intervals the increments of available value for Lander (Figure 4.5.1-2) with the corresponding reliability at the end of that same time interval and plotting these "product" values on the same time scale, the comparison of system effectiveness for various system combinations is provided in Figure 4.5.1-3.

By using the mission value available from each of the two Landers (in the case in which they both survive impact) as noted in Paragraph 4.2.1 A (6), the opportunity is present to acquire the 60 percent of mission value designated for the first Lander on the surface plus the 10 percent designated as the mission value of that Lander's entry data plus the 30 per cent designated for the Orbiter data and in addition acquire the 60 percent of mission value for the surface data from the second Lander plus 5 percent designated for the entry data of the second Lander. Thus, as shown by Figure 4.5.1-2 , the successful landing of a second duplicate Lander at a separate, different site on the planet makes the mission capable of returning more value. This is the same additional value as though the second Lander had been placed there by a separate launch booster and separate interplanetary guidance and control, etc. Thus the upper curve in Figure 4.5.1-2 represents a greater opportunity (165 percent) as compared to any "single Lander surviving" system.

By combining value increments from this upper curve of Figure 4.5.1-2 with corresponding reliabilities (both surviving) of the lower curve of Figure 4.5.1-1, the greater system effectiveness of a two Lander system is shown.

For the purpose of this chart the mission value available from the Orbiter was superposed upon this latter, most effective Lander system. It is, of course, proper to superpose it upon any of the other Lander systems to obtain comparative values of complete systems.

The width of the band shown for each Lander combination in Figure 4.5.1-3 illustrates the effect of variations in terrain reliability (i.e., the probability that the terrain will be suitable for a successful landing at the area of impact as discussed in Paragraph 4.2.1 A (5) over the range of 92 to 98 percent.

The total system effectiveness to be expected of any given launch would be adjusted downward from that shown by the uppermost curve of Figure 4.5.1-3 as a function of the terrain risk and Lander system configuration considered most representative of the particular launch involved.

(5) A plot of Attainable Mission Value at 24 hours was shown in Figure 4. 2. 1-3. This plot was obtained by a study of the relative contribution of each scientific instrument as noted under Table 4.5.1-3 and by multiplying this value increment by its corresponding reliability just as was done in determining the overall system effectiveness in Figure 4.5.1-3 above.

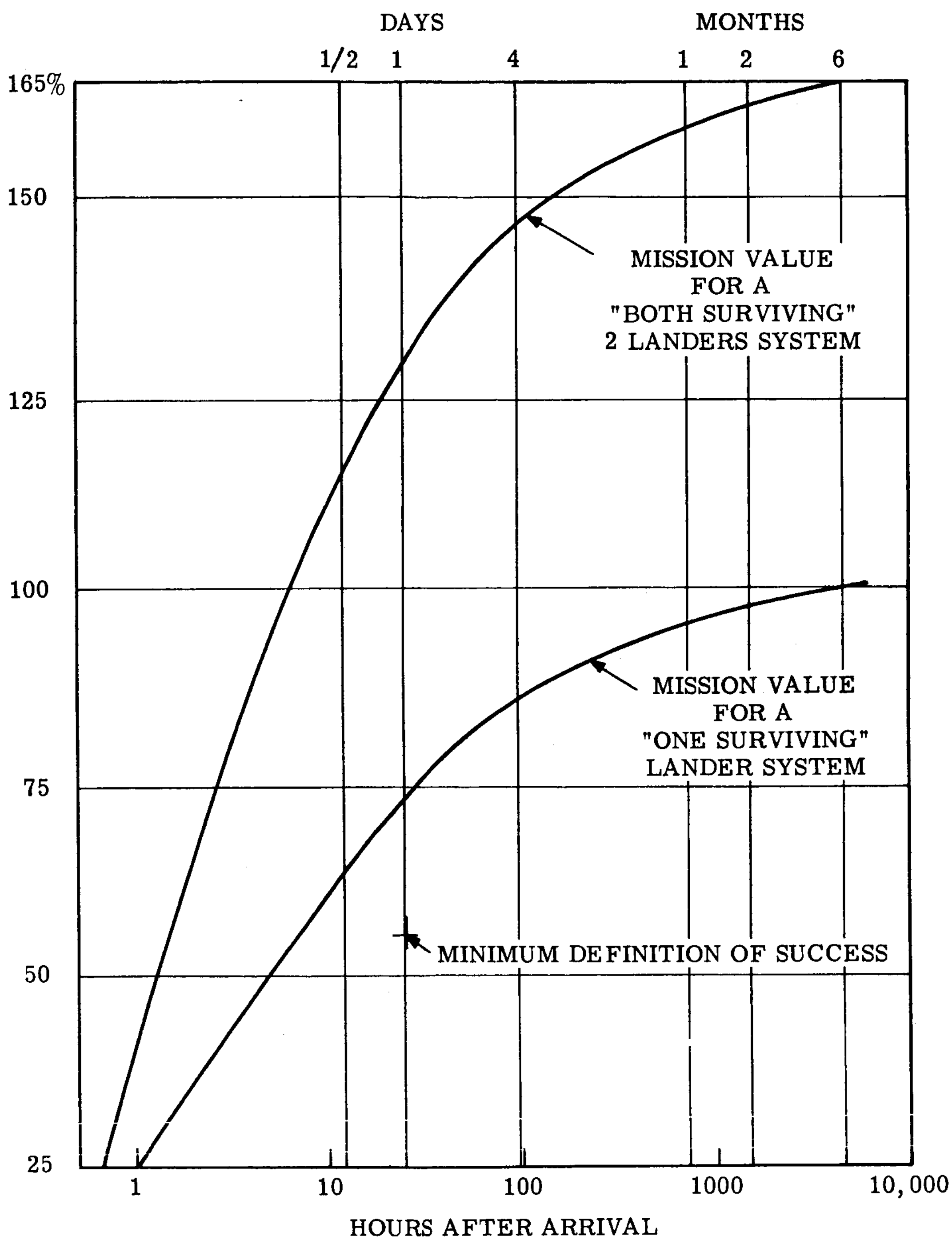


Figure 4.5.1-2 Mission Scientific Value

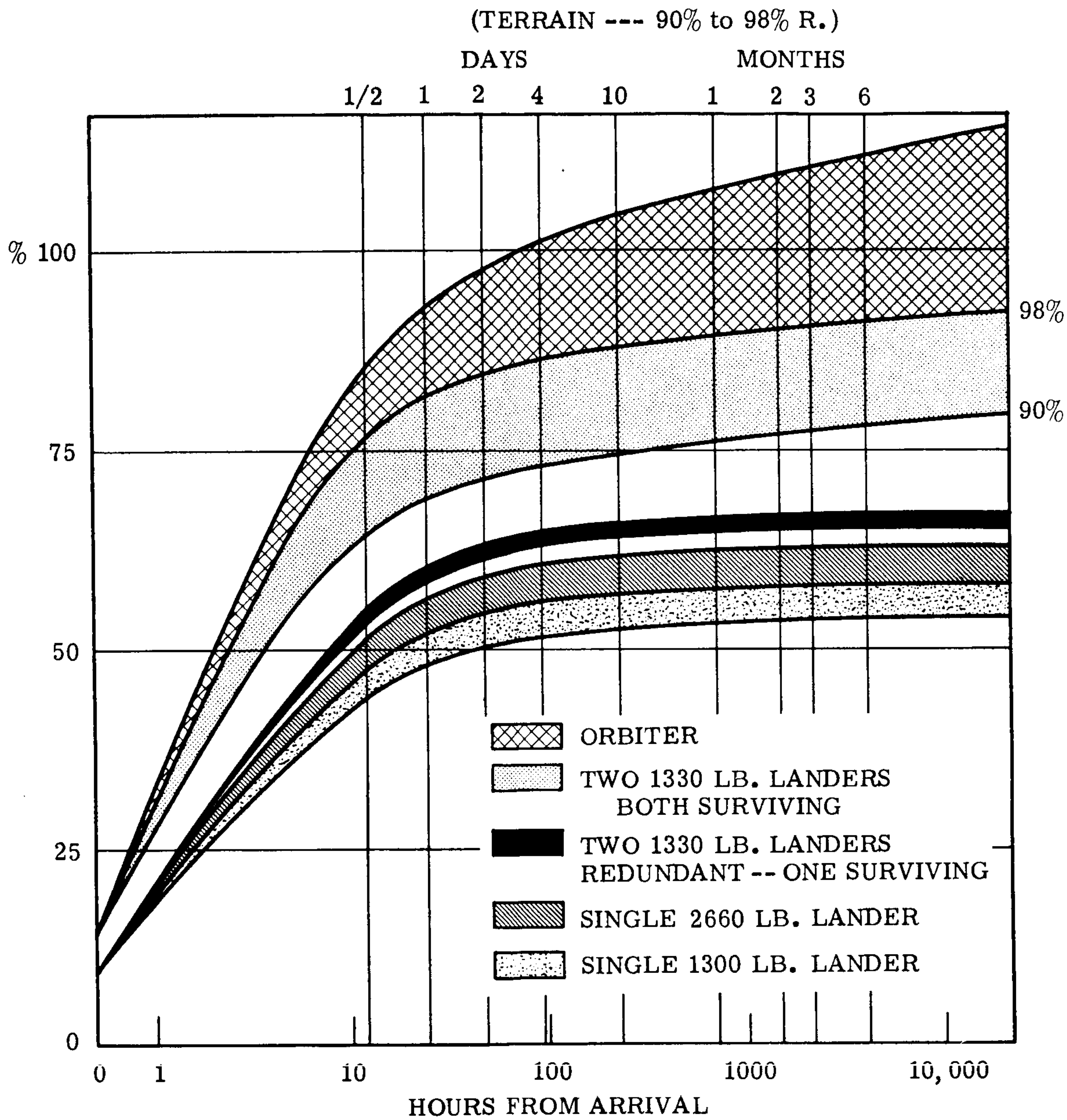


Figure 4.5.1-3 System Effectiveness (Reliability x Mission Value)

TABLE 4.5.1-4a. MISSION VALUE ANALYSIS SHEET

Table 4.5.1-4a Mission Value Analysis Sheet

Priority Per Column 9	MISSION VALUE ANALYSIS SHEET FOR SINGLE LANDER AT 24 HOURS AFTER ARRIVAL WITH: TERRAIN, T, @90% LANDER, R, @84.5% *** INSTRUMENTS, R, @99.5%	1 Instrument Identifying Number	2 Individual Instrument Weight	3 ENTRY: Mission Value -% Available	4 ENTRY: R. R. V. Value -% Attainable	5 SURFACE: Mission Value -% Available @ 24 Hours	6 R. R. V. T. in % = Mission Value Attainable in 24 Hours	7 Cumulative Mission Value Attainable in 24 Hours	8 Cumulative Instrument Weight	9 % Attainable Value Per Pound at 24 Hours
	Name of Instrument		lbs	%	%	%	%	%	lbs	%
1	Temperature	I-24	.3	1.00	.86	1.80	1.34	2.20	.3	7.10
2	Sounds	I-34	.5	-	-	2.55	1.90	4.10	.8	3.80
3	Pressure	I-17	.3	.50	.43	.46	.34	4.87	1.1	2.50
4	Density	I-20	1.5	2.00	1.72	2.40	1.80	8.39	2.6	2.35
5	Multiple Chamber	I-54	4.0	-	-	9.00	6.70	15.09	6.6	1.67
6	Surface Penetration Hardness	I-25	4.5	-	-	1.90	1.42	16.51	11.1	.95**
7	Photoautotroph	I-62	3.0	-	-	2.70	2.01	18.52	14.1	.67
8	Light Intensity (Sun Sensor)	I-84	.5	-	-	.38	.28	18.80	14.6	.58
9	Composition, H_2O	I-44	1.5	.50	.43	.46	.34	19.69	16.1	.51
10	Composition, O_2	I-45	1.5	.50	.43	.46	.34	20.58	17.6	.51
11	Turpidity PH	I-53	4.0	-	-	2.70	2.00	22.58	21.6	.50
12	Wind Speed & Direction	I-67	2.0	-	-	1.20	.90	23.48	23.6	.45
13	Gas Chromatograph	I-8	7.0	2.00	1.72	1.84	1.37	26.57	30.6	.44
14	Composition, N_2	I-48	1.0	.25	.22	.23	.17	26.96	31.6	.39
15	Composition, CO_2	I-49	1.0	.25	.22	.23	.17	27.35	32.6	.39
16	Soil Moisture	I-70	2.0	-	-	.96	.72	28.07	34.6	.36
17	TV Camera, Panorama	TV	20.0*	-	-	9.10	6.80	34.87	54.6	.34
18	Radioisotope	I-19	6.0	-	-	2.70	2.01	36.88	60.6	.33
19	Composition, O_3	I-46	1.5	.25	.22	.23	.17	37.27	62.1	.26
20	Composition, A	I-47	1.5	.25	.22	.23	.17	37.66	63.6	.26
21	Precipitation	I-36	1.0	-	-	.25	.18	37.84	64.6	.18
22	Electron Density (Langmuir Probe)	I-39	3.0	.50	.43	-	-	38.27	67.6	.14
23	Surface Gravity	I-72	3.0	-	-	.50	.37	38.64	70.6	.12
24	Surface Roughness & Altimeter (Pulse Radar)	I-5	15.0	2.00	1.72	-	-	40.36	85.6	.11
25	Microscope, Including TV Camera, Drill, Handling Pulverizer, Sample	I-71	75.0	-	-	8.55	6.37	46.73	160.6	.08
26	Seismic Activity	I-21	8.0	-	-	.35	.26	46.99	168.6	.03
	Lander Subtotals			10.00	8.62	51.18		46.99	168.6*	.15
	Orbiter: 10 Instruments In Order: I-23, 2, 79, TV, 12, 96, 55, 85, 95, 1 Subtotals					15.74		11.44	204.0	.40 to .03
	SYSTEM TOTALS					76.82		58.43	372.6	

* Incl. 10 lbs T V Deployment
** Less 3 lbs deployment
*** Not yet revised to include latest analysis per 4.5.3 A (2)

TABLE 4.5.1-4b. MISSION VALUE ANALYSIS SHEET

Table 4.5.1-4b Mission Value Analysis Sheet

Priority Per Column 9 Table 4.5.1-4a	MISSION VALUE ANALYSIS SHEET FOR SINGLE LANDER AT 96 HOURS AFTER ARRIVAL WITH: TERRAIN, T, @ 90% LANDER, R, @ 84% INSTRUMENTS, R, @ 96.5%	1 Instrument Identifying Number	2 Individual Instrument Weights	3 ENTRY: Mission Value -% Available	4 ENTRY: R. R. V. Value -% Attainable	10 SURFACE: Mission Value -% Available @ 24 Hours to 96 Hours	12 R. R. V. T. in % = Mission Value Attainable @ 24 Hours to 96 Hours	13 Cumulative Mission Value Attainable in 96 Hours	8 Cumulative Instrument Weight	14 % Attainable Value Per Pound at 96 Hours
	Name of Instrument		lbs	%	%	%	%	%	lbs	%
1	Temperature	I-24	.3	1.00	.86	.90	.67	2.87	.3	9.60
2	Sounds	I-34	.5	-	-	.15	.11	4.88	.8	4.02
3	Pressure	I-17	.3	.50	.43	.02	.02	5.67	1.1	2.64
4	Density	I-20	1.5	2.00	1.72	.60	.88	10.07	2.6	2.93
5	Multiple Chamber	I-54	4.0	-	-	.50	.37	17.14	6.6	1.76
6	Surface Penetration Hardness	I-25	4.5	-	-	-	-	18.56	11.1	.95
7	Photoautotroph	I-62	3.0	-	-	.15	.11	20.68	14.1	.71
8	Light Intensity (Sun Sensor)	I-84	.5	-	-	.07	.06	21.02	14.6	.68
9	Composition, H_20	I-44	1.5	.50	.43	.02	.02	22.93	16.1	.53
10	Composition, 0_2	I-45	1.5	.50	.43	.02	.02	23.84	17.6	.53
11	Turpidity & PH	I-53	4.0	-	-	.15	.11	25.95	21.6	.53
12	Wind Speed & Direction	I-67	2.0	-	-	.40	.29	27.14	23.6	.56
13	Gas Chromatograph	I-8	7.0	2.00	1.72	.10	.07	30.30	30.6	.45
14	Composition, N_2	I-48	1.0	.25	.22	.01	.01	30.70	31.6	.40
15	Composition, $C0_2$	I-49	1.0	.25	.22	.01	.01	31.10	32.6	.40
16	Soil Moisture	I-70	2.0	-	-	.01	.01	31.83	34.6	.37
17	TV Camera, Panorama	TV	20.0	-	-	.10	.06	38.69	54.6	.34
18	Radioisotope	I-19	6.0	-	-	.15	.11	40.81	60.6	.35
19	Composition, 0_3	I-46	1.5	.25	.22	.01	.01	41.21	62.1	.27
20	Composition, A	I-47	1.5	.25	.22	.01	.01	41.61	63.6	.27
21	Precipitation	I-36	1.0	-	-	.05	.04	41.83	64.6	.22
22	Electron Density (Langmuir Probe)	I-39	3.0	.50	.43	-	-	42.26	67.6	.14
23	Surface Gravity	I-72	3.0	-	-	-	-	42.63	70.6	.12
24	Surface Roughness & Altimeter (Pulse Radar)	I-5	15.0	2.00	1.72	-	-	44.35	85.6	.11
25	Microscope, Including TV Camera, Drill, Handling Pulverizer, Sample	I-71	75.0	-	-	.47	.35	51.07	160.6	.09
26	Seismic Activity	I-21	8.0	-	-	.05	.04	51.37	168.6	.04
	Lander Subtotals			10.00	8.62	4.40		51.37	168.6	
	Orbiter: 10 Instruments In Order: I-23, 2, 79, TV, 12, 96, 55, 85, 95, 1									
	Subtotals					5.20		15.20	204.	
	SYSTEM TOTALS					86.42		66.57	372.6	

TABLE 4.5.1-4c. MISSION VALUE ANALYSIS SHEET

Priority Per Column 9 Table 4.5.1-4a	MISSION VALUE ANALYSIS SHEET FOR SINGLE LANDER SYSTEM AT 1 MONTH AFTER ARRIVAL WITH: TERRAIN, T = 90% LANDERS, R = 76% INSTRUMENTS, R = 87.5%	1 Instrument Identifying Number	2 Individual Instrument Weights	3 ENTRY: Mission Value -% Available	4 ENTRY: R. R. V. Value -% Attainable	14 SURFACE: Mission Value -% Available @ 96 Hours to 1 Month	15 R. R. V. T. in % = Mission Value Attainable in @96 Hours to 1 Month	16 Cumulative Mission Value Attainable in 1 Month	8 Cumulative Instrument Weight	17 % Attainable Value Per Pound at 1 Month	18 Cumulative Mission Value Attainable in 3 Months
	Name of Instrument		lbs	%	%	%	%	%	lbs	%	%
1	Temperature	I-24	.3	1.00	.86	.15	.09	2.96	.3	9.87	
2	Sounds	I-34	.5	-	-	.15	.12	5.09	.8	4.26	
3	Pressure	I-17	.3	.50	.43	.01	.01	5.89	1.1	2.67	
4	Density	I-20	1.5	2.00	1.72	.20	.12	10.41	2.6	3.02	
5	Multiple Chamber	I-54	4.0	-	-	.40	.24	17.72	6.6	1.82	
6	Surface Penetration Hardness	I-25	4.5	-	-	.06	.04	19.18	11.1	.97	
7	Photoautotroph	I-62	3.0	-	-	.12	.07	21.37	14.1	.73	
8	Light Intensity (Sun Sensor)	I-84	.5	-	-	.04	.03	21.74	14.6	.74	
9	Composition, H_2O	I-44	1.5	.50	.43	.01	.01	22.66	16.1	.53	
10	Composition, O_2	I-45	1.5	.50	.43	.01	.01	23.58	17.6	.53	
11	Turpidity & PH	I-53	4.0	-	-	.12	.07	25.76	21.6	.54	
12	Wind Speed & Direction	I-67	2.0	-	-	.20	.12	27.07	23.6	.66	
13	Gas Chromatograph	I-8	7.0	2.00	1.72	.04	.02	30.25	30.6	.46	
14	Composition, N_2	I-48	1.0	.25	.22	.01	.01	30.66	31.6	.41	
15	Composition, CO_2	I-49	1.0	.25	.22	-	-	31.06	32.6	.40	
16	Soil Moisture	I-70	2.0	-	-	.01	.01	31.80	34.6	.37	
17	TV Camera, Panorama	TV	20.0	-	-	.10	.06	38.72	54.6	.35	
18	Radioisotope	I-19	6.0	-	-	.12	.07	40.91	60.6	.36	
19	Composition, O_3	I-46	1.5	.25	.22	.01	.01	41.32	62.1	.27	
20	Composition, A	I-47	1.5	.25	.22	-	-	41.72	63.6	.27	
21	Precipitation	I-36	1.0	-	-	.10	.06	42.00	64.6	.28	
22	Electron Density (Langmuir Probe)	I-39	3.0	.50	.43	-	-	42.43	67.6	.14	
23	Surface Gravity	I-72	3.0	-	-	-	-	42.80	70.6	.12	
24	Surface Roughness & Altimeter (Pulse Radar)	I-5	15.0	2.00	1.72	-	-	44.52	85.6	.11	
25	Microscope, Including TV Camera, Drill, Handling Pulverizer, Sample	I-71	75.0	-	-	.38	.23	51.47	160.6	.09	
26	Seismic Activity	I-21	8.0	-	-	.08	.05	51.82	168.6	.04	
	Lander Subtotals			10.00	8.62	2.32	1.42	51.82	168.6		51.69
	Orbiter: 10 Instruments In Order: I-23, 2, 79, TV, 12, 96, 55, 85, 95, 1										
	Subtotals					4.57		18.04	204.		19.80
	SYSTEM TOTALS					93.31		69.86	372.6		71.49

TABLE 4.5.1-4d. MISSION VALUE ANALYSIS SHEET

Priority Per Column 9 Table 4.5.1-4a	MISSION VALUE ANALYSIS SHEET FOR DUAL LANDER (BOTH SURVIVING) AT 24 HOURS WITH: TERRAIN $T^2 = (90\%)^2$ LANDERS $R^2 = (84.5\%)^2$ INSTRUMENTS $R^2 = (99.5\%)^2$	1 Instrument Identifying Number	2 Individual Instrument Weights	19 200% of Instrument Weight	20 1.5 $(RR)^2V$ for Entry Values Attainable	21 2.0 $(RRT)^2V$ for Surface Values Attainable at 24 Hours	22 Cumulative Attainable R. V. @ 24 Hours	23 Cumulative 200% of Instrument Weight
	Name of Instrument		lbs	lbs	%	%	%	lbs
1	Temperature	I-24	.3	.6	1.11	2.00	3.11	.6
2	Sounds	I-34	.5	1.0	-	2.84	5.95	1.6
3	Pressure	I-17	.3	.6	.56	.51	7.02	2.2
4	Density	I-20	1.5	3.0	2.22	2.68	11.92	5.2
5	Multiple Chamber	I-54	4.0	8.0	-	10.00	21.92	13.2
6	Surface Penetration Hardness	I-25	4.5	9.0	-	2.12	24.04	22.2
7	Photoautotroph	I-62	3.0	6.0	-	3.00	27.04	28.2
8	Light Intensity (Sun Sensor	I-84	.5	1.0	-	.42	27.46	29.2
9	Composition, H_2O	I-44	1.5	3.0	.56	.51	28.53	32.2
10	Composition, O_2	I-45	1.5	3.0	.56	.51	29.60	35.2
11	Turpidity & PH	I-53	4.0	8.0	-	2.98	32.58	43.2
12	Wind Speed & Direction	I-67	2.0	4.0	-	1.34	33.92	47.2
13	Gas Chromatograph	I-8	7.0	14.0	2.22	2.04	38.18	61.2
14	Composition, N_2	I-48	1.0	2.0	.28	.25	38.71	63.2
15	Composition CO_2	I-49	1.0	2.0	.28	.25	39.24	65.2
16	Soil Moisture	I-70	2.0	4.0	-	1.07	40.31	69.2
17	TV Camera, Panorama	TV	20.0	40.0	-	10.10	50.41	109.2
18	Radioisotope	I-19	6.0	12.0	-	3.00	53.41	121.2
19	Composition, O_3	I-46	1.5	3.0	.28	.25	53.66	124.2
20	Composition, A	I-47	1.5	3.0	.28	.25	53.91	127.2
21	Precipitation	I-36	1.0	2.0	-	.26	54.17	129.2
22	Electron Density (Langmuir Probe)	I-39	3.0	6.0	.56	-	54.73	135.2
23	Surface Gravity	I-72	3.0	6.0	-	.55	55.28	141.2
24	Surface Roughness & Altimeter (Pulse Radar)	I-5	15.0	30.0	2.22	-	58.50	171.2
25	Microscope, Including TV Camera, Drill, Handling Pulverizer, Sample	I-71	75.0	150.0	-	9.50	68.00	321.2
26	Seismic Activity	I-21	8.0	16.0	-	.39	68.39	337.2
	Lander Subtotals				11.13		68.39	337.2
	Orbiter: 10 Instruments In Order: I-23, 2, 79, TV, 12, 96, 55, 85, 95, 1							
	Subtotals						11.44	204.
	SYSTEM TOTALS						79.83	541.2

TABLE 4.5.1-4e. MISSION VALUE ANALYSIS SHEET

Priority Per Column 9 Table 4.5.1-4a	MISSION VALUE ANALYSIS SHEET FOR DUAL LANDER* SYSTEM AT 96 HOURS WITH: TERRAIN $T^2 = (90\%)^2$ LANDERS $R^2 = (84\%)^2$ INSTRUMENTS $R^2 = (96.5\%)^2$	1 Instrument Identifying Number	2 Individual Instrument Weights	19 200% of Instrument Weight	24 1.5 $(RR)^2V$ for Entry Values Attainable	25 2.0 $(RRT)^2V$ for Surface Values Attainable @ 24 to 96 Hours	26 Cumulative R. V. @ 96 Hours	23 Cumulative 200% of Instrument Weight
	Name of Instrument		lbs	lbs	%	%	%	lbs
1	Temperature	I-24	.3	.6	1.11	.98	4.09	.6
2	Sounds	I-34	.5	1.0	-	.16	7.09	1.6
3	Pressure	I-17	.3	.6	.56	.03	8.19	2.2
4	Density	I-20	1.5	3.0	2.22	1.28	14.37	5.2
5	Multiple Chamber	I-54	4.0	8.0	-	.54	24.91	13.2
6	Surface Penetration Hardness	I-25	4.5	9.0	-	-	26.03	22.2
7	Photoautotroph	I-62	3.0	6.0	-	.16	29.19	28.2
8	Light Intensity (Sun Sensor	I-84	.5	1.0	-	.09	29.70	29.2
9	Composition, H_2O	I-44	1.5	3.0	.56	.03	30.80	32.2
10	Composition, O	I-45	1.5	3.0	.56	.03	31.90	35.2
11	Turpidity & PH	I-53	4.0	8.0	-	.16	35.04	43.2
12	Wind Speed & Direction	I-67	2.0	4.0	-	.42	36.80	47.2
13	Gas Chromatograph	I-8	7.0	14.0	2.22	.10	41.16	61.2
14	Composition, N_2	I-48	1.0	2.0	.28	.02	41.71	63.2
15	Composition, CO_2	I-49	1.0	2.0	.28	.02	42.26	65.2
16	Soil Moisture	I-70	2.0	4.0	-	.02	43.35	69.2
17	TV Camera, Panorama	TV	20.0	40.0	-	.10	53.55	109.2
18	Radioisotope	I-19	6.0	12.0	-	.16	56.71	121.2
19	Composition, O_3	I-46	1.5	3.0	.28	.02	57.26	124.2
20	Composition, A	I-47	1.5	3.0	.28	.02	57.81	127.2
21	Precipitation	I-36	1.0	2.0	-	.06	58.13	129.2
22	Electron Density (Langmuir Probe)	I-39	3.0	6.0	.56	-	58.69	135.2
23	Surface Gravity	I-72	3.0	6.0	-	-	59.24	141.2
24	Surface Roughness & Altimeter (Pulse Radar)	I-5	15.0	30.0	2.22	-	61.46	171.2
25	Microscope, Including TV Camera, Drill, Handling Pulverizer, Sample	I-71	75.0	150.0	-	.51	71.47	321.2
26	Seismic Activity	I-21	8.0	16.0	-	-	71.86	337.2
	Lander Subtotals				11.13		71.86	337.2
	Orbiter: 10 Instruments In Order: I-23, 2, 79, TV, 12, 96, 55, 85, 95, 1							
	Subtotals						15.20	204.
	SYSTEM TOTALS						87.06	541.2

*With Both Surviving

TABLE 4.5.1-4f. MISSION VALUE ANALYSIS SHEET

Priority Per Column 9 Table 4.5.1-4a	MISSION VALUE ANALYSIS SHEET FOR DUAL LANDER SYSTEM (BOTH SURVIVING) AT 1 MONTH WITH: TERRAIN $T^2 = (90\%)^2$ LANDERS $R^2 = (76\%)^2$ INSTRUMENTS $R^2 = (87.5\%)^2$	1 Instrument Identifying Number	2 Individual Instrument Weights	19 200% of Instrument Weight	27 1.5 $(RRT)^2V$ for Entry Values Attainable	28 2.0 $(RRT)^2V$ for Surface Values Attainable @ 96 Hours to 1 Month	29 Cumulative R. V. @ 1 Month	23 Cumulative 200% of Instrument Weight	30 Cumulative R. V. @ 6 Months
	Name of Instrument		lbs	lbs	%	%	%	lbs	%
1	Temperature	I-24	.3	.6	1.11	.11	4.20	.6	
2	Sounds	I-34	.5	1.0	-	.14	7.34	1.6	
3	Pressure	I-17	.3	.6	.56	.01	8.35	2.2	
4	Density	I-20	1.5	3.0	2.22	.14	14.67	5.2	
5	Multiple Chamber	I-54	4.0	8.0	-	.29	25.50	13.2	
6	Surface Penetration Hardness	I-25	4.5	9.0	-	.05	27.67	22.2	
7	Photoautotroph	I-62	3.0	6.0	-	.08	30.91	28.2	
8	Light Intensity (Sun Sensor)	I-84	.5	1.0	-	.04	31.46	29.2	
9	Composition, H_20	I-44	1.5	3.0	.56	.01	32.57	32.2	
10	Composition, 0_2	I-45	1.5	3.0	.56	.01	33.68	35.2	
11	Turpidity & PH	I-53	4.0	8.0	-	.08	36.90	43.2	
12	Wind Speed & Direction	I-67	2.0	4.0	-	.14	38.80	47.2	
13	Gas Chromatograph	I-8	7.0	14.0	2.22	.02	43.18	61.2	
14	Composition, N_2	I-48	1.0	2.0	.28	.01	43.74	63.2	
15	Composition, $C0_2$	I-49	1.0	2.0	.28	-	44.29	65.2	
16	Soil Moisture	I-70	2.0	4.0	-	.01	45.39	69.2	
17	TV Camera, Panorama	TV	20.0	40.0	-	.07	55.66	109.2	
18	Radioisotope	I-19	6.0	12.0	-	.08	58.74	121.2	
19	Composition, 0_3	I-46	1.5	3.0	.28	.01	59.30	124.2	
20	Composition, A	I-47	1.5	3.0	.28	-	59.85	127.2	
21	Precipitation	I-36	1.0	2.0	-	.07	60.24	129.2	
22	Electron Density (Langmuir Probe)	I-39	3.0	6.0	.56	-	60.80	135.2	
23	Surface Gravity	I-72	3.0	6.0	-	-	61.35	141.2	
24	Surface Roughness & Altimeter (Pulse Radar)	I-5	15.0	30.0	2.22	-	63.57	171.2	
25	Microscope, Including TV Camera, Drill, Handling Pulverizer, Sample	I-71	75.0	150.0	-	.28	73.86	321.2	
26	Seismic Activity	I-21	8.0	16.0	-	.06	74.31	337.2	
	Lander Subtotals				11.13		74.31	337.2	74.41
	Orbiter: 10 Instruments In Order: I-23, 2, 79, TV, 12, 96, 55, 85, 95, 1								
	Subtotals						18.04	204.	19.80
	SYSTEM TOTALS						92.35	541.2	94.21

TABLE 4.5.1-4g. MISSION VALUE ANALYSIS SHEET

Priority Per Column 9 Table 4.5.1-4a	MISSION VALUE ANALYSIS SHEET FOR DUAL LANDER SYSTEM (ONE SURVIVING) $R = 1-(1-RRT)^2$	1 Instrument Identifying Number	2 Individual Instrument Weights	19 200% of Instrument Weight	31 1.25 (Column 7)* for Cumulative Value Attainable in 24 Hours	32 1.27 (Column 13)** for Cumulative Value Attainable in 96 Hours	33 1.4 (Column 16)*** for Cumulative Value Attainable in 1 Month	23 Cumulative 200% of Instrument Weight	34 Cumulative Value Attainable in 6 Months
	Name of Instrument		lbs	lbs	%	%	%	lbs	%
1	Temperature	I-24	.3	.6	2.76	3.64	4.14	.6	
2	Sounds	I-34	.5	1.0	5.14	6.18	7.11	1.6	
3	Pressure	I-17	.3	.6	6.12	7.19	8.24	2.2	
4	Density	I-20	1.5	3.0	10.52	12.74	14.58	5.2	
5	Multiple Chamber	I-54	4.0	8.0	18.95	21.7	24.8	13.2	
6	Surface Penetration Hardness	I-25	4.5	9.0	20.75	23.5	26.8	22.2	
7	Photoautotroph	I-62	3.0	6.0	23.2	26.0	29.9	28.2	
8	Light Intensity (Sun Sensor)	I-84	.5	1.0	23.6	26.6	30.4	29.2	
9	Composition, H_2O	I-44	1.5	3.0	24.7	29.0	31.7	32.2	
10	Composition, O_2	I-45	1.5	3.0	25.6	30.2	33.0	35.2	
11	Turpidity & PH	I-53	4.0	8.0	28.4	32.9	36.0	43.2	
12	Wind Speed & Direction	I-67	2.0	4.0	29.5	34.4	37.9	47.2	
13	Gas Chromatograph	I-8	7.0	14.0	33.4	38.4	42.3	61.2	
14	Composition, N_2	I-48	1.0	2.0	33.9	38.9	42.9	63.2	
15	Composition, CO_2	I-49	1.0	2.0	34.3	39.4	43.5	65.2	
16	Soil Moisture	I-70	2.0	4.0	35.2	40.3	44.5	69.2	
17	TV Camera, Panorama	TV	20.0	40.0	43.8	49.0	54.1	109.2	
18	Radioisotope	I-19	6.0	12.0	45.3	51.6	57.2	121.2	
19	Composition, O_3	I-46	1.5	3.0	46.9	52.2	57.8	124.2	
20	Composition, A	I-47	1.5	3.0	47.3	52.7	58.4	127.2	
21	Precipitation	I-36	1.0	2.0	47.5	53.0	58.8	129.2	
22	Electron Density (Langmuir Probe)	I-39	3.0	6.0	48.0	53.5	59.4	135.2	
23	Surface Gravity	I-72	3.0	6.0	48.5	54.1	59.9	141.2	
24	Surface Roughness & Altimeter (Pulse Radar)	I-5	15.0	30.0	49.5	56.1	62.4	171.2	
25	Microscope, Including TV Camera, Drill, Handling Pulverizer, Sample	I-71	75.0	150.0	58.6	64.7	72.0	321.2	
26	Seismic Activity	I-21	8.0	16.0	59.9	65.0	72.5	337.2	
	Lander Subtotals				58.0	65.0	72.5	337.2	73.5
	Orbiter: 10 Instruments In Order: I-23, 2, 79, TV, 12, 96, 55, 85, 95, 1 Subtotals				11.4	15.2	18.1	541.2	19.8
	SYSTEM TOTALS				69.4	70.2	90.6		93.3

*See Table 4.5.1-4a
**See Table 4.5.1-4b
***See Table 4.5.1-4c

Figure 4.5.1-4 shows these same Attainable Mission Value curves. In addition, using the left hand vertical scale, the net payload weight carrying capability of a single Lander is plotted vs the overall weight of the entry Lander. This line is used to establish what instruments could be carried by an entry Lander of any given payload carrying capability.

Each step in the superposed Attainable Mission Value curve represents the addition of another instrument. Each adds its increment of value to compose the cumulative total mission value attainable with a single Lander of the corresponding overall entry Lander weight. In preparing this chart, the Attainable Mission Value of each instrument at various points in time (24 hours, 96 to 100 hours, 1 month, etc.) was calculated. The instrument having the greatest Attainable Mission Value per pound of instrument weight was given first priority in being placed aboard a Lander; and so on, in this same order of priority for the other instruments. The methods used and the Tables of values obtained are reported below in Paragraph 4.5.1 A (7) and Table 4.5.1-4.

The net scientific payload weight carrying capability of two identical Landers is twice that of a single Lander. The overall weight of two Landers is twice also. Thus, the smooth curve on the right side of Figure 4.5.1-4 represents the total payload capability of two Landers.

Correspondingly, each step of the Attainable Mission Value curve for Dual Landers represents the weight and value of two identical instruments, one in each Lander and these are applied in the same order of priority as discussed above to insure the most effective use is made of all available scientific payload weight. (Any other order could be used but would, of course, represent a comparison of non-optimum systems.) However, should the scientist's evaluation of the relative value of any instrument be modified as the result of a comparable instrument being available in the other Lander (recognizing the risk that it may not survive and perform), any other optimum combination of instruments could be used with the two Landers differing then in this respect.

Thus the 1840 pound point above which two Landers of the design proposed by this study are preferable to any single Lander system considered is graphically shown by Figure 4.5.1-4.

(6) Either or both of the two Landers might also be modified with respect to communications, power supply, etc. as well as in its scientific instrument complement.

To illustrate the effects of such alternatives, the possibility of reducing the communication rate capability of the Lander to the Orbiter from the 7000 bits per second of the proposed design to 1/10th that or 700 bits per second was briefly investigated. This indicated that approximately 100 lbs could be saved in the combined Power Supply and Communications subsystems area per Lander. The effect of such a saving has an influence on the weights of other subsystems. The results of this inquiry are indicated in Figure 4.5.1-5. In such a system, the point at which two (2) Landers offer definite advantage over a single Lander system is considerably lower (1200 pounds) then in Figure 4.5.1-5. It should be carefully noted that this 700 bit/sec rate was not adopted in the final design proposed by this study and is included here simply as illustrative of the usefulness and flexibility of this method.

Figure 4.5.1-6 shows the subsystem composition of Lander Vehicle weight as function varies total vehicle weights. In arriving at these curves a number of specific designs were reviewed. The distributions were determined for these designs and the curves were obtained by interpolation. The shape of the curves is considerably influenced by the logarithmic scale used for Lander vehicle weight. The bottom curve showing NET INSTRUMENT WEIGHT available in a Lander which also provides a 7000 bit per second communication capability via the Orbiter is the same line as is shown for the single Lander in Figure 4.5.1-4.

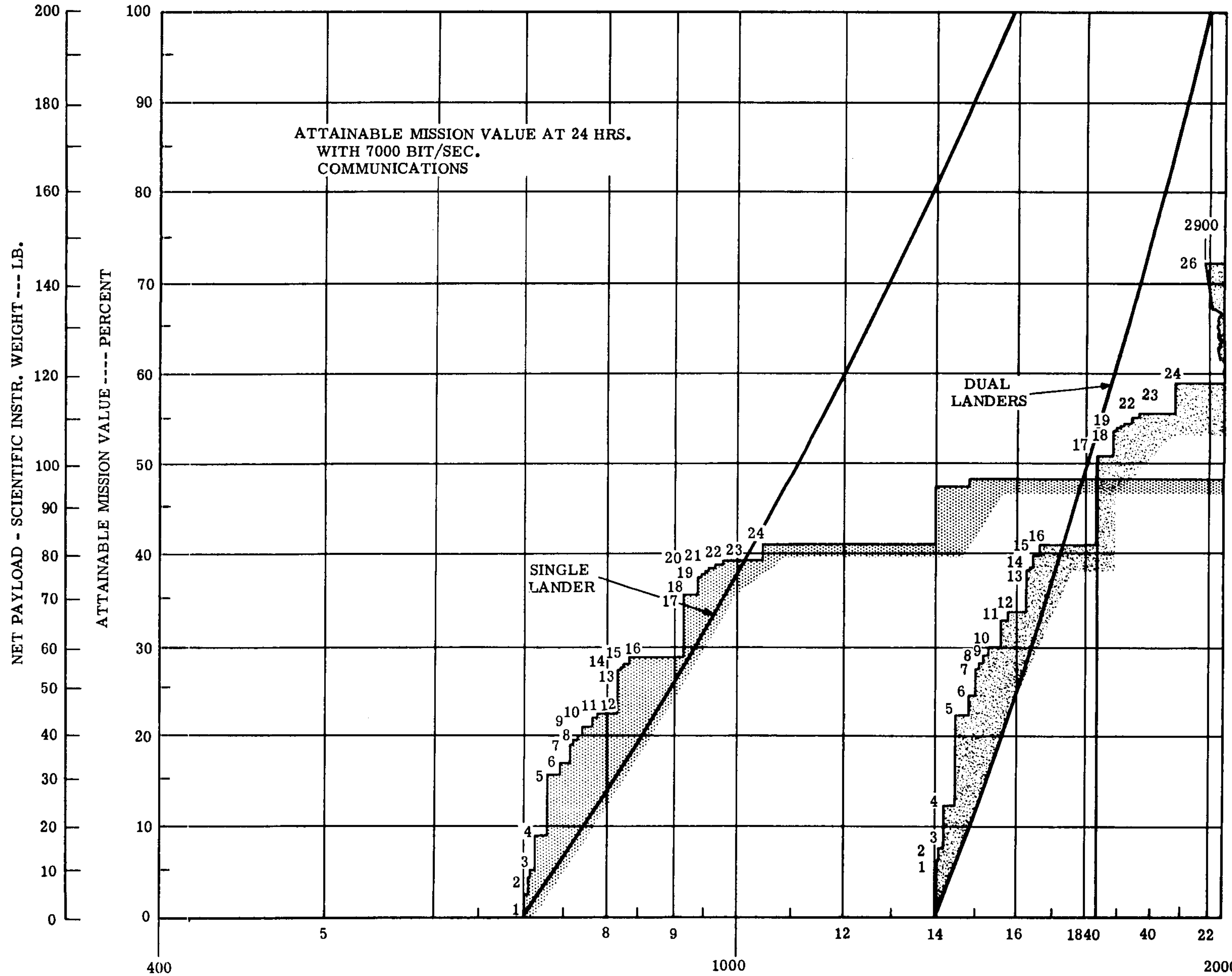


Figure 4.5.1-4. Single vs. Dual Lander

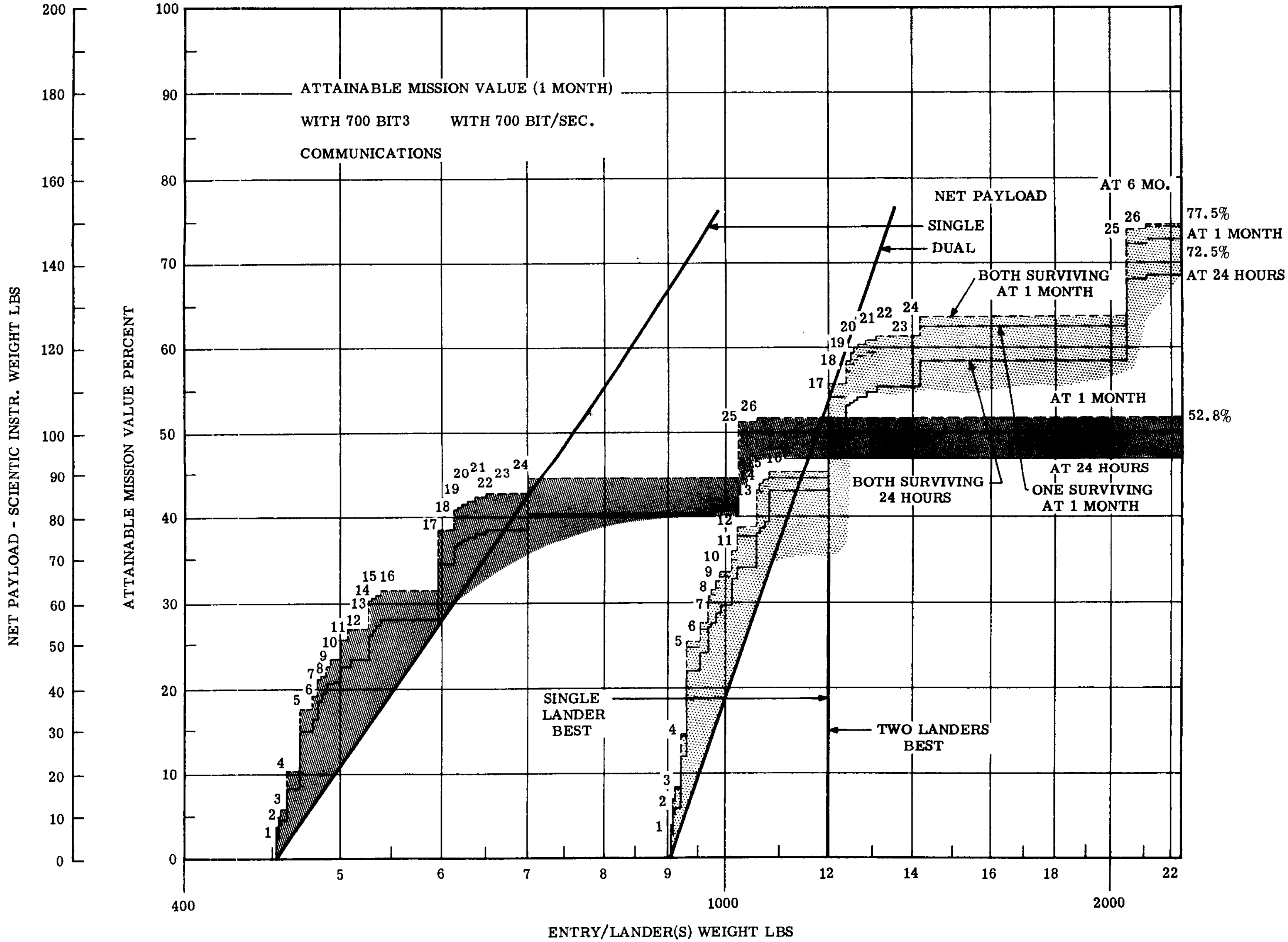


Figure 4.5.1-5. Percent Increase in Lander Reliability Using Communication Vis Direct Link vs Via Orbiter To Earth

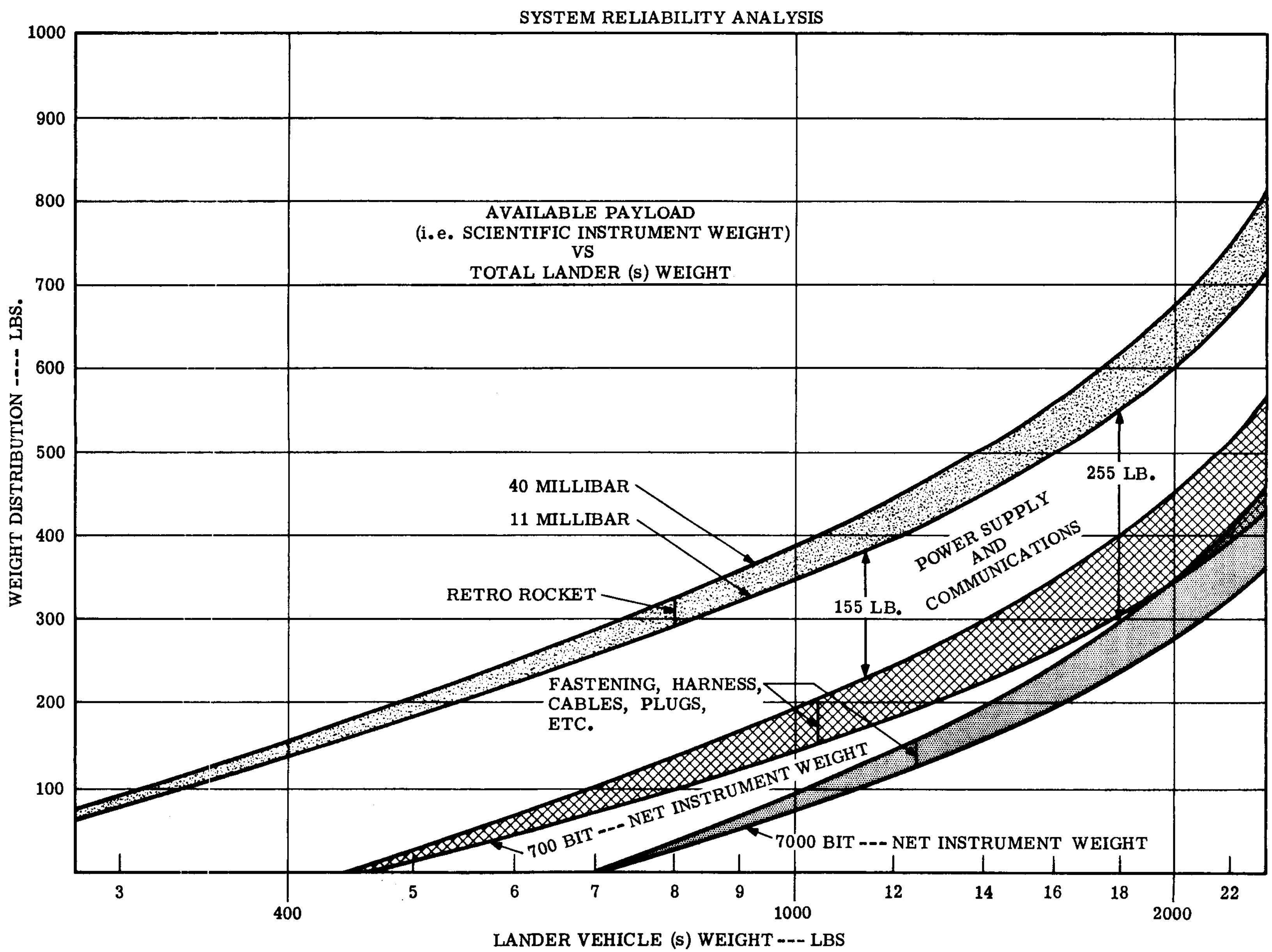


Figure 4.5.1-6. System Reliability

Also, the lower edge of the next higher curve shows the NET INSTRUMENT WEIGHT available in a Lander which had 100 lb less devoted to power supply and communications and which provides an approximate 700 Bit/Sec communication capability. This is the same line as is shown for the Net Payload — Single Lander in Figure 4.5.1-5.

(7) Method of Approach for Systems Reliability and Trade-off Analysis for:

1. Single vs dual Landers — system optimization

2. Lander instruments vs Orbiter — system optimization

3. Overall mission systems cost and reliability trade-off analysis — system optimization.

(a) Single vs Dual Lander

For two or more Landers arriving at significantly separate locations on a planet, the Mission Value available to each instrument of each Lander is considered to be equal to that available to an identical instrument on a large, single Lander at either location. For atmospheric entry data, the Mission Value available to like instruments on successive Landers is considered to be one-half of that available upon the preceding Lander.

1. Determine the reliabilities of each component of the overall system and optimize the system design (less scientific instruments) to determine the available scientific instrument weight and maximum overall mission value.

2. Determine gross weight of Landers combined.

3. Determine gross weight of each Lander.

4. Determine net scientific instruments weight for each Lander.

5. Determine the weight of each scientific instrument including any special devices or deployment mechanisms peculiarly required by the individual instrument.

6. Determine the cumulative Mission Value in percent[1] available to a single, large Lander.

7. Select suitably significant periods (if desired, these periods may be reduced in size until a complete integral of attainable Mission Value is obtained) of time after the arrival of each Lander and establish the available Mission Value for each instrument during each of these periods.

   This is done by weighing the judgments made of the relative contributions of each instrument to the individual mission's objectives and apportioning the Mission Value of the large single Lander to them.

8. Determine the cumulative Mission Value in percent[1] available to each of the alternative Landers.

---

[1] Where the Mission Value available (i.e., were 100 percent Reliability applicable) through the application of the gross Lander weight to a single Lander plus the Mission Value available from the Orbiter or fly by vehicle(s) is considered equal to 100 percent and where the portion of this value attributed to the single large Lander has been specified.

9. For the duration of each of those periods, determine the reliability (i. e., the sum of probabilities of successful operation at the beginning and at the end of this period divided by two) applicable to each instrument. This reliability includes:

   a. R_L – The reliability of the Lander at the beginning and at the end of the period, including its means of communication via the Orbiter (or direct) to the receiving station on the earth and the cumulative effects of prior environmental and operational periods from the time of launch.

   b. T – The reliability of impact including the probability that the local terrain in combination with atmospheric densities, entry trajectories, etc., is suitable (i. e., within the "design limits" established for the Lander design and development) to the Lander.

   c. R_S – The reliability of the scientific instrument, including all factors as in (a) above.

10. Determine the Mission Value attainable by each instrument during each time period (i. e., the product of the Available Value and the applicable reliability for that period).

11. For the total mission period and any significant period (item 7 above) establish the Attainable Mission Value Per Pound(2) for each instrument (i. e., item 9 divided by the corresponding item 4) and arrange the list of instruments in order of decreasing Attainable Mission Value per pound.

12. Apply(2) instruments to each Lander in the sequence of "greatest Attainable Mission Value per pound" first and thereafter in order of decreasing value.

13. Summarize the resultant Attainable Mission Values per Lander and per system of alternative Lander configurations and combinations. Review(2) these against the applicable Mission Objectives and iterate any revisions in:

   a. Instrument selection and evaluation per item 6, etc., above

   b. Available total scientific instrument weight

   c. System and component reliabilities

   d. Available Mission Value for each instrument

   to optimize the overall systems design to obtain maximum overall Attainable Mission Value. From this summary determine this maximum Attainable Mission Value in percent for the total mission period.

(b) Orbiter vs Lander(s)

1. Apply the same procedure, as outlined in steps 1 to 10 under (a) Single vs Dual Lander to establish the Attainable Mission Value per pound for each instrument carried by the Orbiter.

---

(2) Any improvements in overall Attainable Mission Value which result from scientific re-evaluations of the relative value per instrument and application must be iterated through each of the preceding steps affected by it.

2. Apply the instruments to theOrbiter in the sequence of "greatest Attainable Mission Value per pound" first and thereafter in order of decreasing value.

3. Summarize the resultant Attainable Mission Values per Orbiter and per system of alternative Orbiter configurations and combinations. Review(2) these against the applicable Mission Objectives and iterate any revisions in:

   a. Instrument selection and evaluation

   b. Available total scientific instrument weight

   c. System and component reliabilities

   d. Available Mission Value for each instrument with particular attention to the values given to Orbiter vs single large Lander instruments to optimize the combined Orbiter and Lander(s) system design to obtain maximum overall Attainable Mission Value. From this summary determine this maximum Attainable Mission Value in percent for the total mission period.

(c) Overall Mission-Systems Cost and Reliability Trade-Off Analyses

1. Determine the cost of each scientific instrument involved in the Mission (including that of any instruments applicable to the mission but which were eliminated by weight restrictions) and the costs of all major elements of the overall mission complex including:

   a. The overall R&D and production costs per Voyager Vehicle system per launch

   b. The overall support operations cost per launch (including DSIF, etc.)

   c. The overall R&D and production costs per Booster (as well as other expendable systems) per launch.

2. Determine the costs of alternative combinations of Booster systems, Landers, etc. by which the mission objectives are considered to be obtainable. Data per item 1 above, even though the accuracy of certain portions may be of preliminary estimates, should be established for each significant combination.

3. Determine the reliability applicable to each mission system combination per launch including all portions of (b) and (c) of item 1 above. Also determine (d), the reliability of attaining two successful mission launches with the designated support facilities during the available launch period (i. e., launch window) based upon the numbers of launch systems which can be fired during that period and their reliabilities.

4. Determine that Attainable Mission Value for the complete Mission System by multiplying that of the Voyager Vehicle system (item 3 under (b) Orbiter vs Lander) by the reliabilities of item 3 (b), (c), and (d) above.

(8) Significant factors contributing to the reliability of the system are presented in section 4.3. These factors are discussed in turn in the individual subsystem analyses.

The subsystem analyses do not detail the subsystem operations, since the technical subsystem discussions cover this area.

B. MATHEMATICAL MODEL

Mathematical models are used to describe the contribution of each functional equipment's success probability relative to the overall system success probability. The reliability values employed in the model for the functional elements are based on the variables associated with a particular mission use of the equipment.

All models used in this analysis are based on the assumption of the exponential distribution since the failure distributions of electronic equipment in the time domain generally exhibit the characteristics of this distribution.

All reliability values were estimated from:

1. Duty cycle of individual components in the mission
2. Estimated parts complexity of each component
3. Thermal control to maintain ambient part case temperatures
4. Partial or complete redundancy, where applied.

The system model for the Voyager Mars 1969 Mission shows the Landers in redundancy throughout both the transit and separation-landing phases. Although the Landers are non-contributory during the transit phase, they are in redundancy, since the failure of any one in this interval would still leave one operational in the following phases.

Mathematical Model:

Probability of success of Voyager 1969 Mars Mission for 100 hours and 3 months after Lander Separation:

$$R\,(\text{system}) = R(\text{Orbiter}) \cdot \left[1-(1-R\ \text{Lander})^2\right]$$

Substituting computed reliability values in the above equation —

$$\underset{\text{100 Hours}}{R(\text{system})} = (.730)\left[1-(1-\ .850)^2\right]$$

$$= (.71)$$

$$\underset{\text{3 Months}}{R(\text{system})} = (.59)$$

For a summary of Voyager Mars 1969 System reliability see Table 4.5.1-5.

Estimated system reliability 3 months after separation

$$R_{(\text{system})} = R_{(\text{Orbiter})} \left[1-(1-R_{\text{Lander}})^2\right]$$

$$= (.618) \qquad (.959)$$

$$= (.59)$$

TABLE 4.5.1-5. MARS 1969

VOYAGER SYSTEM

RELIABILITY SUMMARY

Orbiter			Lander			
Subsystem	Reliability 100 Hr.	Reliability 3 Month	Subsystem	Reliability 100 Hr.	Reliability 3 Month	Reliability 6 Month
Communication	.836	.761	Communication	.998	.962	.921
Guidance & Control	.898	.838	EP&D	.963	.951	.940
Power Supply	.984	.980	Prop. & Sep.	.968	.968	.968
Propulsion			Thermal Control	.945	.932	.923
Hot Gas	.999	.999	Retardation	.986	.986	.986
Cold Gas	.99	.99	Orientation	.981	.981	.981
Orbiter Vehicle Reliability	.730	.618	Lander Vehicle Reliability	.850	.798	.748
			Redundant Landers	.978	.959	.937

Estimated system reliability 100 hours after separation

$$
\begin{aligned}
R_{(system)} &= R_{(Orbiter)} \left[1 - (1 - R_{Lander})^2\right] \\
&= (.730)\,(.978) \\
&= (.71)
\end{aligned}
$$

C. OPERATIONAL STATES

During the Voyager mission all equipments are either fully energized, cycled, or in the off state, according to the sequence in which their function is required. Thus, the parts and circuits within the equipment are subjected to various degrees of stress, relative to the operational state they are in.

Recognizing that the lifetime of a part is a function of the stress level and the interval of the applied stress, modifying factors are employed herein to account for the operational state of the parts throughout the mission.

As a reference point, one hour of continuous operation in the normal interplanetary space environment is used as a base and assigned a value of unity, K = 1. Other modifying factors used are:

Launch Phase K = 100

Off State K = .015

1st Hour of
cyclic state K = 4

These factors are primarily used in the launch and transit phases where periodic diagnostic data transmission or reorientation of the spacecraft for on-course correction is required.

Table 4.5.1-6 illustrates the sequence of events during the interplanetary transit period where diagnostic and scientific monitoring information may be required. The effect of the modifying factors on the actual operate time of the components results in an "effective" operational time greater than, equal to, or less than the base value.

TABLE 4. 5. 1-6. COMBINED SCHEDULE

TABLE 4. 5. 1-6. COMBINED SCHEDULE

CUMULATIVE REAL TIME-DAYS	1. Voyager Sequence of Events – Mars 1969 2. Monitoring 3. Reliability Effects of On-Off Cycles and of "OFF" time during transit	CUMULATIVE EFFECTIVE HOURS FOR LANDER	Minimum Schedule For Mars 1969 EFFECTIVE HOURS FOR ORBITER	CUMULATIVE "ON" HOURS	CUMULATIVE EFFECTIVE HOURS FOR ORBITER
0	Launch, K = 100 (K = 1 in space)	10	.1	.1	10
	Entry into Transit Mode		6. 5	6. 6	16. 6
	(30 Sets Diag. Data Incl. in A)		–		
1	1 Set Diag. Data/Hr. for 1st Day		17. 5	24	34
	2 Sets Diag. Data/Day for 1st Week		12	36	46
7	Effective T = 4 Hours Per Start		(48)		94
	1 Set Diag. Data/Day for 2nd Week		7	43	101
14	**T = 4 x 7 Starts**		(28)		129
	2 Sets Data/Week for 3rd Week		2	45	131
	T - 4 x 2 Starts		(8)		139
	Monitor of G&C, Etc.		–		
	Start Gyros & Monitor		1	46	140
	T = 4 x 1 Start		(4)		144
20	First Midcourse Maneuver		1	47	145
21	Reorientation and Monitor of G&C, Etc.		1	48	146
28	2 Sets of Data in 4th Week		2	50	148
	T = 4 x 2 Starts		(8)		156
35	2 Sets of Data in 5th Week		2	52	158
	T = 4 x 2 Starts		(8)		166
	Monitor of G&C, Etc.		–		
	Start Gyro's & Monitor		1	53	167
	T = 4 x 1 Start		(4)		171
	Second Midcourse Maneuver		1	55	173
	Reorientation & Monitor		1	62	180
	1 Set Data/Wk – Next 7 Weeks		7	62	180
84	T = 4 x 7 Starts		(28)		208
	1 Set Data/Wk – Next 29 Weeks		29	91	237
	T = 4 x 29 (Incl. TG)		(116)		353
	Monitor of C&C, Etc.		–		
187	High-Gain Antenna Deployment @ 4500 hours		1	92	354
	Terminal Guidance Observation		1	93	355
276	Final Trajectory Correction		2	95	357
	Reorientation and Monitoring		2	97	359
279	Orbiter/Lander Ejection (150,00 N. M.) to Entry		2	99	361
	Orbiter Reorientation to the Sun	19. 5	1	100	362
	T = 4 x 1 Start		(4)		366
	Orbiter Retardation				
	Orbiter Reorientation and Orbit Injection		6	106	372
	T = 4 x 1 Start		(4)		376
	Midcourse & Monitoring Contingency		14	120	390
	Effective Hours of Orbiter "Off" Time (Mars)		(6)		
280	(6960-120) 1. 5/100 = 104		(104)		500
	Lander Entry to Impact	20			
	Lander Impact Konvir 1 - Min. 600	(10)			
	Lander "Off" Time Effective				
	6950 x 1. 5/100 = 105	(105)			
	End of Transit Period			120	
	Use for Mars	135			500
	19 Fewer Weeks of Transit Time for Venus	(-50)	(-50)		
	Use for Venus	85			450

Table 4. 5. 1-7 shows failure rates during "off" time/"on" time as a ratio (%) of "on" time λ .

TABLE 4. 5. 1-7. FAILURE RATES DURING "OFF" TIME/"ON" TIME AS A RATIO (%) OF "ON" TIME λ **

PART TIME	"OFF"	"1ST HOUR ON"
Vacuum Tubes	1/100th to 1/1,000th	25 Times
Unpassivated Semiconductors	1/10th to 1/100th	10 to 20 Times
Passivated Semiconductors	1/100th to 1/1,000th	2 to 4 Times
Resistors	1/100th to 1/10,000th	1 to 4 Times
Capacitors	1/100th to 1/10,000th	1 to 4 Times
Inductive Devices	1/100th to 1/10,000th	1 to 4 Times
Connections	1/100th to 1/10,000th	1 to 4 Times
Relays, Switches, Potentiometers	1/100th to 1/10,000th	1 to 4 Times
Motors, Rockets, Squibs Lead Styphanate) up to Lead Azide) 3 years	1/100th to 1/10,000th	1
Seals Valves, Fittings, Tanks, Pumps, Bearings, & All other Such Items	1/100th to 1/10,000th	1

*Every Individual Usage Must be Justified vs High Failure Rates

** λ = Failure rate, usually in %/1000 hours

The following bit shows the average system composition by "part type failure rate"

1. Less than 10 percent of electronics will apply "Unpassivated" transistors
2. From 40 to 60 percent of electronics, λ , is in semiconductors
3. Over 90 percent of Parts are transistors, diodes, resistors, capacitors.

Using maximum "Start" (1st Hours) rates the electronics subsystems and components will have 5 times λ_o, during each 1st hour after starting. Factors will be lower for non-electronics.

Using maximum "Off" time rates these subsystems and components will have 1. 5/100 times λ_o during each "Off" period. Factors will be lower for non-electronics.

4. 5. 2 ORBITER SYSTEM

A System Definition

The Orbiter system has multiple functions in the mission. During the transit phase it is the Earth-vehicle communications link, performs maneuvers and transmits diagnostic data.

In the orbiting phase it acquires and transmits scientific information to Earth, maintains two way communications with the Landers and exercises stabilization and control of the main vehicle.

To insure the success of these functions, various features have been incorporated into the design of the Orbiter to maintain total uninterrupted or degraded operation in the event of partial or complete component failure.

Three methods are employed to sustain operational continuity (1) complete redundancy of components, (2) internal circuit redundancy (majority logic) or (3) programming of alternate functional loops (stand-by redundancy). The definition of these features and their areas of use is shown below in Table 4. 5. 2-1.

Table 4. 5. 2-1. Methods Employed to Sustain Operational Continuity

Complete Redundancy	Redundancy (Majority Logic)	Stand-By Redundancy
Pitch, Yaw & Roll Amplifiers	Command & Computer Equipment (Comm)	Star-Trackers
	Data Processor (Comm)	Earth Trackers
	Storage & Logic Unit (G&C)	Hi-Gain & Omni
		Hot Gas System

(1) Reliability Analysis

The Orbiter portion of the Voyager vehicle is the only communication link during transit for commands and data transmission. Equipment within the Orbiter is energized periodically by command or pre-programming for maneuvers or diagnostic data transmission.

Therefore, most of the equipment is in the "off state" when not required for functional duty, thereby reducing power requirements and electrical stress on component parts.

Standby redundancy is extensively used in the subsystems, as well as alternate modes of operation. Although the alternate modes do not have the same degree of performance as the primary operational mode, they prevent total loss of a mission function. These alternate modes and redundant features are described in the Orbiter subsystems discussion in conjunction with their effect on the estimated reliability.

(2) Mathematical Model

Mathematical models are shown for the 2 phase Orbiter operation.

$$\underset{100\text{ hrs}}{R_{(\text{Orbiter})}} \quad R_{(\text{power supply})} \cdot R_{(\text{G\&C})} \cdot R_{(\text{Communications})} \cdot R_{(\text{hot gas})} \cdot R_{(\text{cold gas})}$$

$$\underset{100\text{ hrs}}{R_{(\text{Orbiter})}} = (.73)$$

$$R_{\substack{(\text{Orbiter}) \\ 3 \text{ months}}} = (.61)$$

B. Subsystem Definition

(1) Communications (See Figure 4. 5. 2-1)

Four modes of Orbiter communications are provided for particular time phases in the mission for vehicle to Earth or Orbiter-Lander links. See Table 4. 5. 2-2.

Table 4. 5. 2-2. Modes of Orbiter Communication

PHASE	MODE	PRIMARY LOOP	BACKUP
1. Transit 10-4500 Hours	Vehicle to Earth	Omni	Hi-Gain
2. Transit 4500 to end of Orbit	Vehicle to Earth	Hi-Gain	Omni
3. Separation to Impact	Lander-Orbiter Reception only	VHF Omni	
4. Lander Impact to End of Orbit	Orbiter-Lander 2 way link	Yagi	

In addition to the primary loops, a data processing, storage unit and a television subsystem comprise the communications subsystem.

Standby redundancy is provided for in two of the four modes by switching in the alternate backup loop in case of failure in the primary loop. Degraded performance will be experienced but catastrophic failure will be prevented. Table 4. 5. 2-3 shows the estimated reliability data generated from the duty cycles of individual components, component failure rates and the use of the "K" modifiers on operating time.

It should be noted that the stated times reflect the sequential communication links that operate only at specific mission milestones, and that continuous operation for the full mission period of 9170 hours is not required of any component.

The component numbers in the first column of the table are used in the mathematical models for simplicity.

(a) Reliability Analysis

The four sequentially operated communication links are designed to fill the broad spectrum of requirements necessary for a combined Orbiter-Earth link and an Orbiter-Landers link.

Certain design features are incorporated in the subsystem to increase reliability, such as:

1. The duty cycle of components are kept to a minimum by turn-on -off programming or switching techniques.

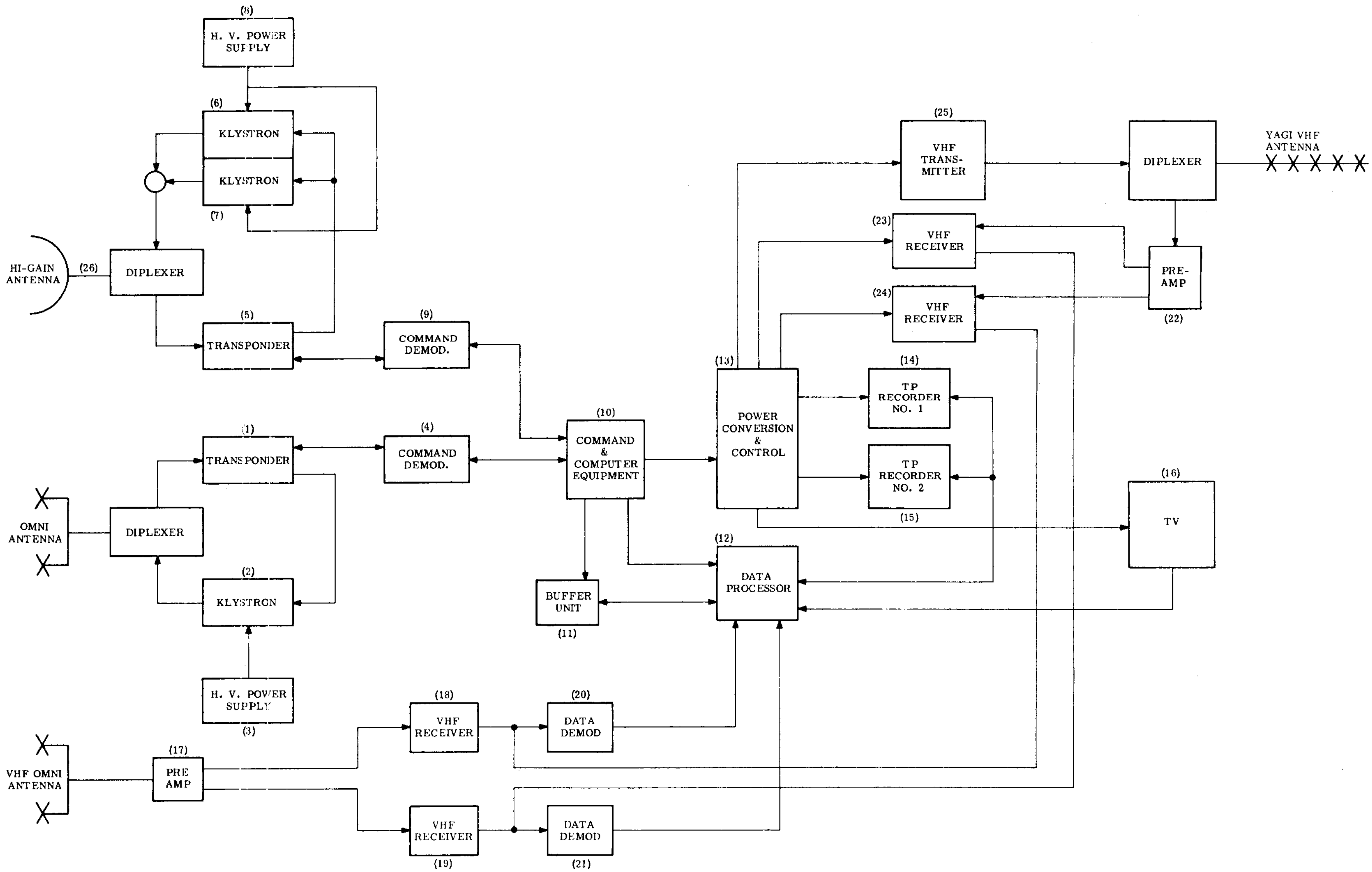


Figure 4.5.2-1 Communications Subsystem - Mars 1969 Mission

TABLE 4.5.2-3. ORBITER COMMUNICATIONS SUBSYSTEM

Comp. No.	Component		Failure Rate %/1000 (hrs)	100 hours Orbit		3 months Orbit	
				Effective Time (hrs)	Rel.	Effective Time (hrs)	Rel.
1	Transponder	(Omni)	1.060	4500	.953	4500	.953
2	Klystron		1.000	410	.999	410	.999
3	H.V. Power Supply		.249	410	.999	410	.999
4	Command Demodulator		.254	4500	.985	4500	.985
5	Transponder	(Hi-Gain)	1.060	2635	.972	4735	.951
6	Klystron		1.000	147	.999	2247	.978
7	Klystron		1.000	147	.999	2247	.978
8	H.V. Power Supply		.249	147	.999	2247	.995
9	Command Demodulator		.254	2635	.994	4735	.988
10	Command & Computer Equip.		.340	2400	.992	4510	.985
11	Buffer Unit		3.500	124	.996	349	.988
12	Data Processor		.698	124	.999	349	.999
13	Power Conversion & Control		.002	Mission	.998	Mission	.998
14	T.P. Recorder		3.180	214	.993	2314	.929
15	T.P. Recorder		3.180	214	.993	2314	.929
16	Image Orthicon		1.256	115	.999	115	.999
17	PreAmp (VHF Omni)		.012	125	.999	125	.999
18	VHF Receiver		.161	125	.999	125	.999
19	VHF Receiver		.161	125	.999	125	.999
20	Data Demodulator		.254	140	.999	840	.999
21	Data Demodulator		.254	140	.999	840	.999
22	PreAmp (VHF Yagi)		.012	215	.999	2315	.999
23	VHF Receiver		.161	215	.999	2315	.996
24	VHF Receiver		.161	215	.999	2315	.996
25	VHF Transmitter		.088	215	.999	2315	.998
26	Hi-Gain Antenna & Diplexer		1.820	2635	.953	4735	.917

Note: All antennae and diplexers not listed in the above Table are considered to have a reliability of approximately 1.0 due to extremely low failure rates.

2. Majority logic will be used in the logic circuitry.

3. Only the receiver circuits of the transponders will be energized during the transit phase.

4. Standby redundancy is used in the Hi-Gain loop with dual klystrons as backup.

5. The data processor and buffer unit will be energized approximately 2-4 hours a day in the orbit phase.

6. The omni VHF loop is only in operation during the separation-Lander-impact phase of the mission.

7. Both the thermoplastic recorders will only be required when a high rate of data acquisition is necessary.

(b) Mathematical Models

The mathematical models define the components in each functional loop, the backup capability and the mathematical interaction of the components.

$$R_{(\text{Orbiter Communications Separation + 100 Hours})} = R_{(\text{Omni Loop})} \cdot R_{(\text{HI-Gain Loop})} \cdot R_{(\text{VHF Omni Loop})} \cdot R_{(\text{Yagi VHF Loop})} \cdot R_{(\text{TV})} \cdot R_{(\text{Data Conversion})}$$

$$= (.937)\,(.919)\,(.995)\,(.995)\,(.999)\,(.984)$$

$$= .836$$

$$R_{(\text{Orbiter Communications Separation +3 Months})} = (.937)\,(.856)\,(.995)\,(.988)\,(.969)\,(.999)$$

$$= (.761)$$

Where

$$R_{(\text{Omni loop})} = (R_1\ R_2\ R_3\ R_4)$$

$$R_{(\text{Hi-Gain Loop})} = (R_5\ R_6\ (1+\lambda t)\ R_8\ R_9\ R_{26})$$

$$R\ (\text{VHF Omni Loop}) = R_{17}\ R_{18}\ R_{19}\ R_{20})$$

$$R_{(\text{Yagi VHF Loop})} = R_{22}\ R_{23}\ R_{24}\ R_{25})$$

$$R_{(\text{Data Conversion})} = R_{10}\ R_{11}\ R_{12}\ R_{13}\ R_{14}\ (1+\lambda t)$$

$$R\ (\text{TV}) = R_{16}$$

Alternate Modes – Stand-by Redundancy (Backup) in Communications S/S.

1. Start of transit phase to 4500 hours where omni loop is primary means of communication with hi-gain in standby redundancy.

Mathematical Model:

$$R_{(\text{communications up to 4500 hrs})} = R_{(\text{omni loop})} \frac{\lambda\,(\text{hi-gain loop})}{\lambda\,(\text{hi-gain} - \lambda\,(\text{omni loop})} \left[R_{(\text{omni loop})} - R_{(\text{hi-gain loop})} \right]$$

2. 4500 hours until mission completion where hi-gain loop is primary, with omni loop in stand-by redundancy.

Mathematical Model:

$$R_{\substack{\text{(communications)} \\ 4500 + \text{hrs}}} = R_{\substack{\text{(hi-gain} \\ \text{loop)}}} \frac{\lambda \text{ (hi-gain loop)}}{\lambda \substack{\text{(omni} \\ \text{loop)}} - \lambda \substack{\text{(hi-gain} \\ \text{loop)}}} \left[R_{\substack{\text{(hi-gain} \\ \text{loop)}}} - R_{\substack{\text{(omni)} \\ \text{loop}}} \right]$$

(2) Guidance and Control

The Guidance and Control Subsystem is designed to perform:

1. Transit orientation
2. Inertial reference
3. Antenna pointing
4. Orbit orientation
5. PHP pointing.

Its three functional areas are: (See Block Diagrams, Figures 4. 5. 2-2 and 4. 5. 2-3)

1. Attitude Control
2. Earth Tracker and Antenna Drive
3. PHP Axes Control.

Attitude Control furnishes fine attitude correction to the vehicle by the magnitude of the error signals received from attitude sensors in the pitch, yaw and roll axes.

Attitude Control is furnished by firing coupled cold gas jets. The firing time is dependent on the magnitude of the error signals received from attitude sensors in the pitch, roll and yaw axes.

The earth tracker and antenna drive keep the communications antenna pointed to the earth by means of an earth sensor and a gimbaled antenna.

The PHP axis control uses the planet sensor to point the gimbaled package to the planet for VHF communication.

(a) Reliability Analysis

Because attitude corrections will be necessary throughout the entire mission, the high usage equipments required for this function are in total redundancy or an alternate mode of operation is provided, given that a failure occurs in the primary mode.

All amplifiers, pitch, yaw and roll, are in redundancy and the earth tracker can be used as a backup to the star tracker during some parts of the mission.

In the transit phase, the major sensing elements are in continuous operation whereas the gyros and the other components only have periodic operation for monitoring purposes or reorientation. (See sequence of events in Table 4. 5. 1-6.)

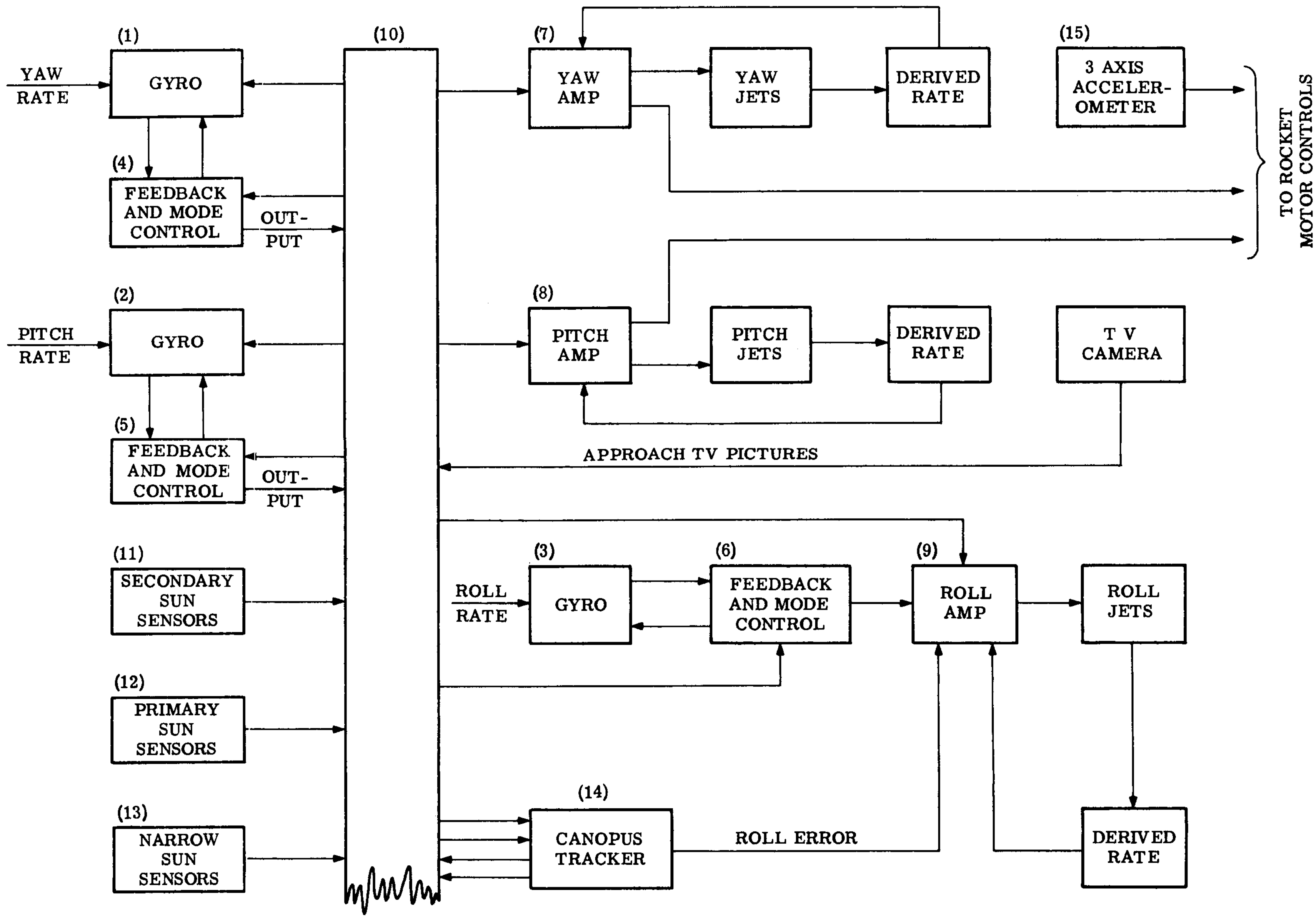


Figure 4.5.2-2 Voyager Guidance and Control Attitude Control

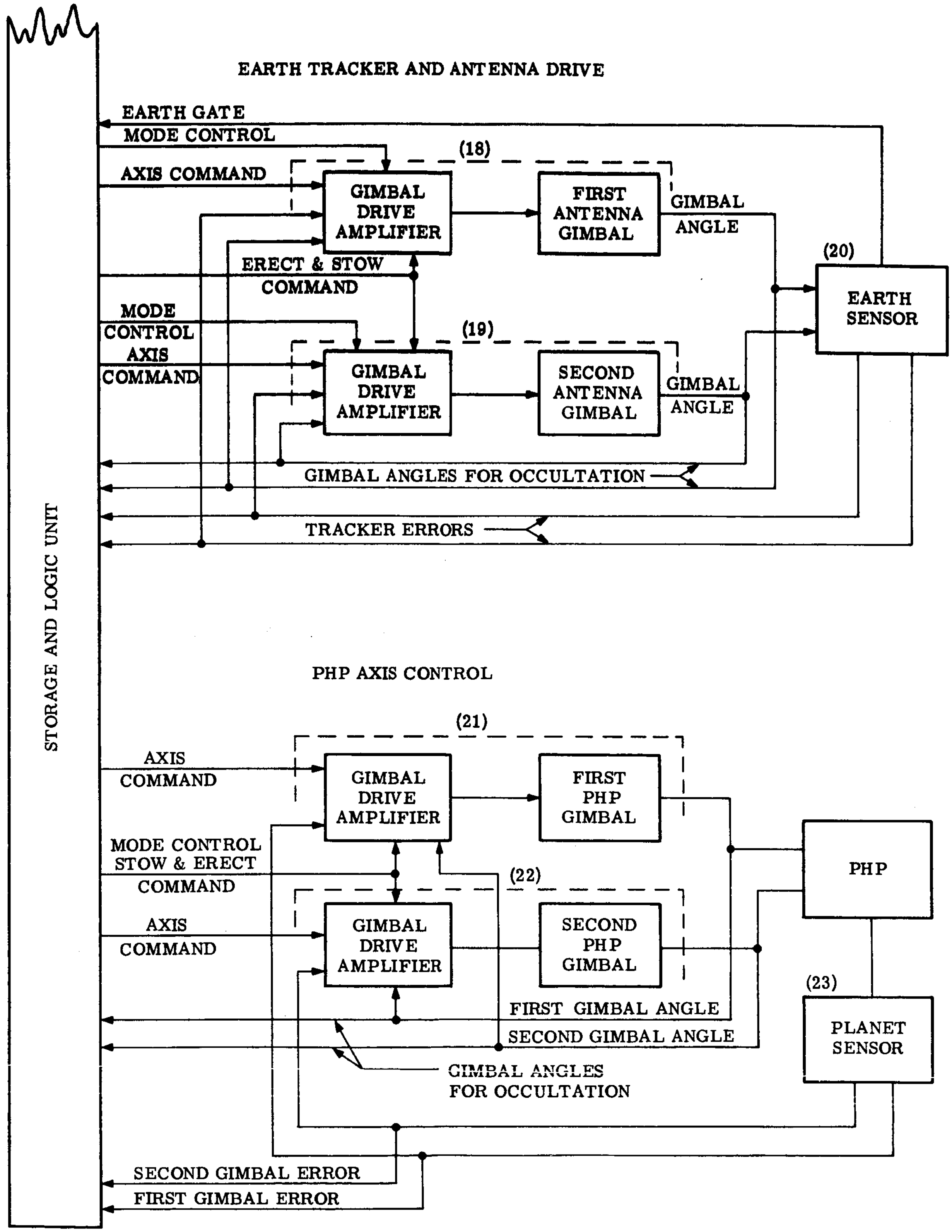


Figure 4.5.2-3 Voyager Guidance and Control

In the orbiting phase, all components except the gyros are assumed to be in continuous operation.

The failure of the narrow sun sensor degrades orientation function, but the vehicle then relies on the primary sun sensor for orientation,

The storage and logic unit has internal circuit redundancy and the majority of the circuits will only "see" a 60 percent duty cycle in the mission.

All gyros have a lifetime requirement of 8000 hours, whereas the estimated use time in the Mars 1969 Mission is ~200 hours.

The thrust vector control and accelerometer are expected to have an operational life of only 20 hours, since they will only be energized prior to and during any hot gas firing.

The PHP package and planet sensor is only operational from orbit injection onward.

(b) Mathematical Models (see Table 4.5.2-4)

The mathematical models for the Guidance and Control Subsystem show the components that are required to operate throughout the entire mission, and the backup modes available in case of a functional failure of the primary mode of operation.

$$R_{\substack{\text{(Orbiter)}\\ \text{G\&C}\\ \text{100 hours}\\ \text{\& 3 months}}} = R_{\substack{\text{(star}\\ \text{tracker)}}} \cdot R_{\substack{\text{(narrow sun}\\ \text{sensor)}}} \cdot R_{\text{(primary sun sensor)}}$$

$$\cdot R_{\substack{\text{(secondary}\\ \text{sun sensor)}}} \cdot \left[1 - (1-R_{\text{amplifier}})^2\right]^3$$

$$\cdot R_{\substack{\text{(storage \&}\\ \text{logic unit)}}} \cdot R^3_{\text{(gyros)}} \cdot R_{\text{(earth sensor)}}$$

$$\cdot R^2_{\text{(antenna servos)}} \cdot R^2_{\text{(PHP servos)}}$$

$$\cdot R^3_{\substack{\text{(feedback \& mode}\\ \text{control amplifiers)}}} \cdot R^2_{\substack{\text{(thrust vector}\\ \text{control)}}}$$

$$\cdot R_{\text{(accelerometer)}} \cdot R_{\text{(planet sensor)}}$$

$$R_{(100 \text{ hrs})} = (.982)(.999)(.994)(.999)\left[1-(1-.994)^2\right]^3$$

$$(.981)(.999)^3(.979)(.988)^2(.999)^2(.999)^3$$

$$(.999)^2(.999)(.999)$$

$$= (.898)$$

$$R_{(3\ mos.)} = (.977)\ (.999)\ (.993)\ (.999)\ \left[1-(1-.992)^2\right]^3$$

$$(.976)\ (.998)^3\ (.964)\ (.978)^2\ (.989)^2$$

$$(.999)^3\ (.999)^2\ (.999)\ (.996)$$

$$= (.838)$$

Alternate backup mode:

Earth tracker in standby redundancy to star tracker:

$$R_{(star\ tracker)} = \left[R_{(star\ tracker)}\ \frac{\lambda\ (star\ tracker)}{\lambda\ (earth\ tracker) - \lambda\ (star\ tracker)}\left(R_{(star\ tracker)} - R_{(earth\ tracker)}\right)\right]$$

TABLE 4.5.2-4. ORBITER GUIDANCE AND CONTROL SUBSYSTEM

Comp. No.	Component	Failure Rate %/1000 hrs	100 Hours Orbit Effective Time (hrs)	100 Hours Orbit Rel.	3 Months Orbit Effective Time (hrs)	3 Months Orbit Rel.
1	Gyro (yaw)	.500	170	.999	350	.998
2	Gyro (pitch)	.500	170	.999	350	.998
3	Gyro (roll)	.500	170	.999	350	.998
4	Feedback & Mode Cont. (yaw)	1.200	170	.999	350	.999
5	Feedback & Mode Cont. (pitch)	1.200	170	.999	350	.999
6	Feedback & Mode Cont. (roll)	1.200	170	.999	350	.999
7	Power Amplifier (yaw)	.093	7070	.994	9170	.992
8	Power Amplifier (pitch)	.093	7070	.994	9170	.992
9	Power Amplifier (roll)	.093	7070	.994	9170	.992
10	Storage & Logic Unit	.440	4260	.981	5520	.976
11	Secondary sun sensors	.080	160	.999	160	.999
12	Primary sun sensors	.080	7070	.994	9170	.993
13	Narrow sun sensors	.010	7070	.999	9170	.999
14	Star Tracker	.256	7070	.982	9170	.977
15	Accelerometer	.178	90	.999	120	.999
16	Thrust vector control	.228	90	.999	120	.999
17	Thrust vector control	.228	90	.999	120	.999
18	Antenna Servo (first)	.468	2600	.988	4700	.978
19	Antenna Servo (second)	.468	2600	.988	4700	.978
20	Earth Sensor	.797	2600	.979	4700	.964
21	PHP Servo (first)	.468	215	.999	2315	.989
22	PHP Servo (second)	.468	215	.999	2315	.989
23	Planet Sensor	.176	215	.999	2315	.996

(3) Power Supply Subsystem (see Figure 4.5.2-4)

The Voyager Power Supply Subsystem uses silicon solar cells as the primary power source, with a nickel cadmium battery as a backup for peak power loads. A regulator limits the average battery charging current and the maximum voltage imposed on the battery to prescribed nominal values. The regulator will also serve as a battery over voltage control in the event that chemical degradation of the battery allows an overvoltage to exist.

The battery capacity required, maximum depth of discharge and the charge rate has been determined for the 1969 mission on the basis of analytical and empirical considerations.

(a) Reliability Analysis

All components within the subsystem, except the battery, are in continuous usage during the mission. The battery is trickle charged from the solar array, and is estimated to be in use for only the high rates of acquisition (TV observation) during the orbiting phase and for mid-course maneuvering, Lander separation and orbit injection during the transit phase.

(b) Mathematical Model (see Table 4.5.2-5)

$$R_{\substack{(\text{power supply})\\ \text{S/S}}} = R_{(\text{solar array})} \cdot R_{(\text{battery})} \cdot R_{(\text{regulator})}$$

$$R_{(100\text{ hrs})} = (\sim 1.0)\,(.999)\,(.985)$$

$$= .984$$

$$R_{(3\text{ months})} = (\sim 1.0)\,(.999)\,(.981)$$

$$= .980$$

TABLE 4.5.2-5. ORBITER POWER SUPPLY SUBSYSTEM

Comp. No.	Component	Failure Rate %/1000 hrs	100 Hours Orbit		3 Months Orbit	
			Effective Time (hrs)	Rel.	Effective Time (hrs)	Rel.
1	Solar Array	.0001	7070	~1.0	9170	~1.0
2	Regulator	.211	7070	.985	9170	.981
3	Battery	.050	123	.999	295	.999

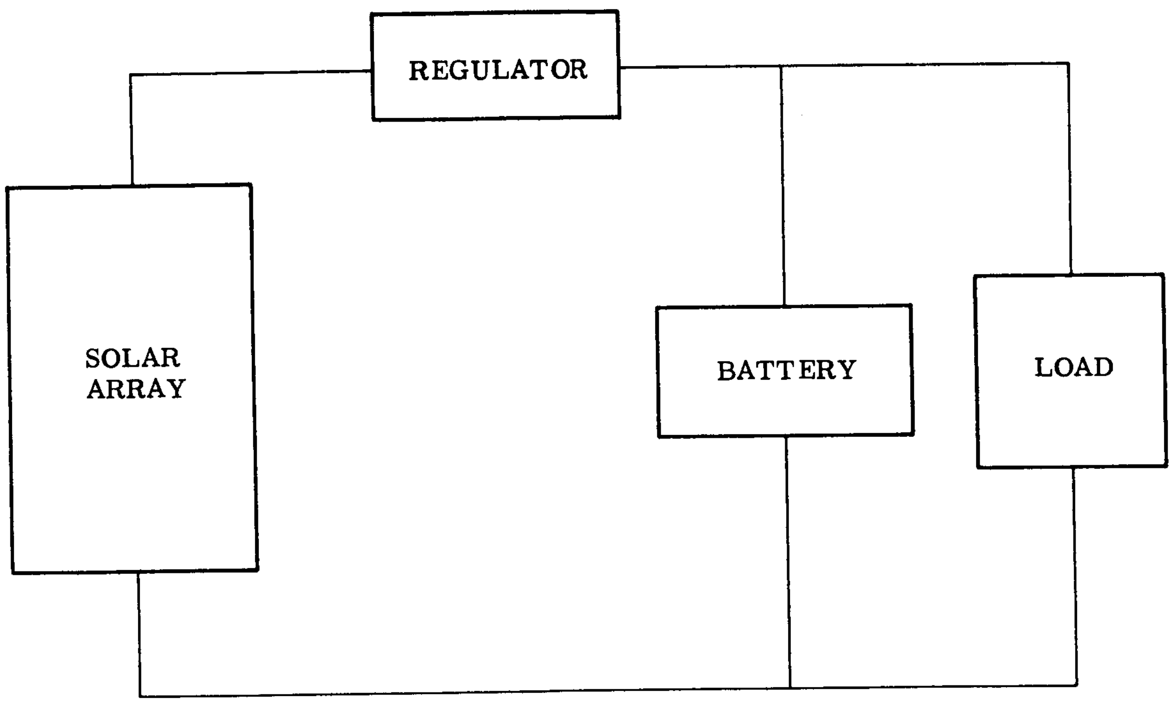


Figure 4.5.2-4 Power System Simplified Block Diagram

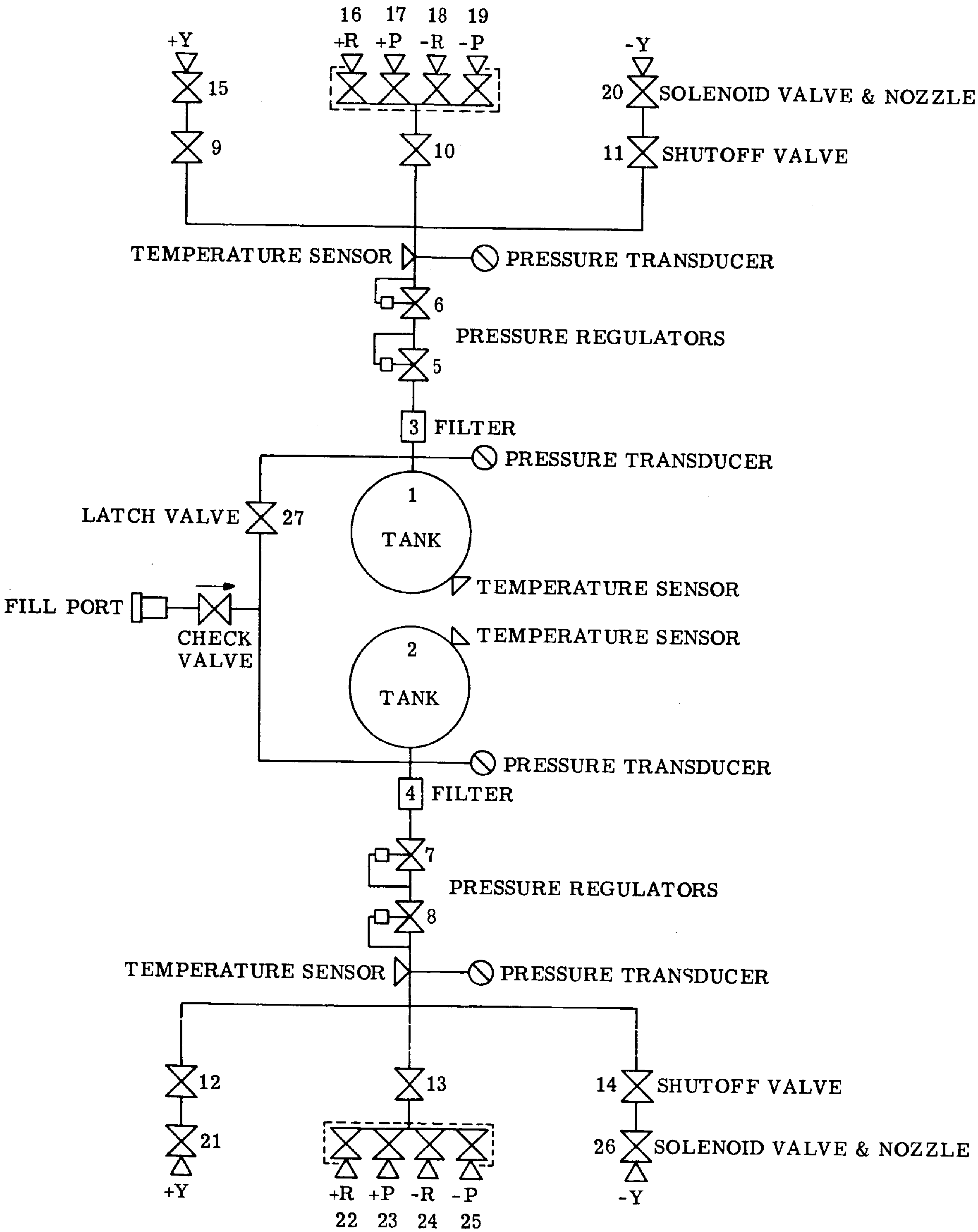


Figure 4.5.2-5. Attitude Control Propulsion Schematic

(4) Propulsion Subsystem

The propulsion subsystem has both a hot gas system for maneuvers and course correction and a cold gas system for attitude control.

The hot gas system is bi-propellant with a command restart capability. A maximum number of 6 start-restarts is estimated for the Mars 1969 mission, with a total burn time of approximately 10 minutes, worst case. All firings of the system will occur in the transit or orbit injection phases.

Due to the extended time interval between firings, the effect of space storage on all components and fuel lines should be investigated to determine if corrosion of fuel lines or hang-up of operating components can occur.

Freon 14 is proposed for use in the cold gas system as the propellant for fine stabilization and control of the vehicle.

Coupled gas jets are used in each axis for + and - control and to retard translation.

(a) Reliability Analysis (Hot Gas System)

The design of the Hot Gas System includes extensive redundancy to prevent the three principal modes of failure normally associated with any fluid system:

1. Leakage
2. Failed open valves
3. Failed close valves

All major functional components are in working or standby redundancy to isolate or bypass those units which may fail in any one of the modes.

Those components which are used for ground fill or system check-out are not functional during powered flight. The potential leakage of these components is prevented by capping after check-out.

Working or standby redundant components are identified below:

1. Working Components
    a. Filters
    b. Regulators
    c. Main pressurizing valves
    d. Solenoid valves
    e. Check valves
    f. Relief valves
2. Standby Components
    a. Propelland and control valves.

Those components not in redundancy are helium gas reservoir, oxidizer and fuel tanks, level detectors, isolation valves, thrust vector nozzles and shut-off valve and thrust chamber.

(b) Mathematical Model

The model does not include the components used only for ground check-out. It shows those components used in any form of redundancy, taken from the system schematic shown in the technical engineering discussion.

$$R_{(\text{Hot Gas System})} = R_{(\text{helium tank})} \cdot \left[1-\left(1-R_{(\text{filter})}\right)^2\right] \cdot \left[1-\left(1-R^2_{(\text{regulator})}\right)\right]^2$$

$$\cdot \left[1-\left(1-R_{(\text{main pressuring valve})}\right)^2\right] \cdot R_{(\text{shut-off valve})} \cdot R_{(\text{oxidizer tank and bellows})}$$

$$\cdot R_{(\text{fuel tank and bellows})} \cdot R^2_{(\text{level detectors})} \cdot \left[1-\left(1-R_{(\text{filter})}\right)^2\right]^2$$

$$\cdot \left[R_{(\text{primary propellant and control valve})} \cdot (1+\lambda t) \cdot\right] \cdot R_{(\text{thrust chamber})}$$

$$\cdot \left[1-\left(1-R_{(\text{normally closed solenoid valves})}\right)^2\right]^2 \cdot \left[1-\left(1-R_{(\text{check valves})}\right)^2\right]^2$$

$$\cdot \left[1-\left(1-R^2_{(\text{relief valves})}\right)^2\right] \cdot \left[1-\left(1-R_{(\text{squibs})}\right)^2\right]^4$$

$$\cdot R^3_{\text{TVC (solenoid valves)}} \cdot R^3_{(\text{nozzles})}$$

where:

$$R_{(\text{primary propellant and control valves})} = 1-\left[1-R_{\text{isolation valve}} \cdot R^2_{\text{solenoid valve}} \cdot R^2_{\text{actuators}} \cdot R^4_{\text{valves}}\right]^2$$

Substituting reliability values obtained from the product of the component failure rates and effective operational time shown in Table 4.5.2-6.

$$R_{\begin{subarray}{l}\text{(Hot Gas}\\ \text{System)}\end{subarray}} = (.999) \cdot 1-(1-.999)^2 \cdot 1-(1-.999)^2 \cdot 1-(1-.999)^2 \, (.999)$$

$$(.999)\,(.999)\,(.999)^2 \cdot \left[1-(1-.999)^2\right]^2 \cdot \left[.999\,(1+.0001)\right]$$

$$(.999)\left[1-(1-.999)^2\right]^2 \cdot \left[1-(1-.999)^2\right]^2 \left[1-(1-.999)^2\right]^2$$

$$\left[1-(1-.999)^2\right]^4 (.999)^3\,(.999)^3$$

Since only 3 significant figures are used in the computations, the estimated reliability could not reflect the true significance of all the redundant features incorporated in the design, therefore, a reliability value of ~ .999 more closely approximates the true reliability and is used in all subsystem and system models for the hot-gas system.

(c) Reliability Analysis (cold gas system) — See Table 4.5.2-7

The reliability of the cold gas system for guidance and control is directly dependent upon the performance life characteristics of the valves selected for final inclusion in the actual system. It is equally dependent upon the influence of the disturbing torques existing in space flight, and the guidance and control programming of the valves established for the final configuration of the operational system.

During the course of this study contract the total number of cycles per valve was expected to be at or below 2500 total cycles. For this type of operation a cold gas subsystem reliability of .99 or greater was determined as applicable to this subsystem and this value has been used in establishing the overall system reliability shown in Figure 4.2.1-6.

Recent reviews of the dynamics of the guidance and control subsystem have indicated that a considerable number of cycles (e.g., up to a maximum of 50,000 cycles per regulator, in the extreme case) may be required. This analysis has not been completed in sufficient depth to resolve this figure in any final sense.

Should such a high number of actuations be required, the use of an inertial system (or equivalent) to greatly reduce the actuations per mission would be required, or an extensive life test program must be instituted for each valve type, in conjunction with a research and development program, to establish valve design with a performance life characteristic > 200,000 open and close cycles.

Leakage from joints, seals or seats is the most common failure occurrence in gas systems. Several design features are included in this system to preclude this type of failure mode, such as:

1. All tanks and pressure transducers are welded units.
2. Dual regulators are used in series so that if either unit hangs open, the other unit will retard leakage.
3. Shut off valves, downstream from the regulators, prevent any one of the solenoid valves from hanging open for excessive time periods, other than firing times.

TABLE 4.5.2-6. ORBITER PROPULSION SUBSYSTEM

Component No.	Component	Quantity	Failure Rate %/1000 hrs	Reliability (Effective Time=116 hrs)
1	Tank, Gas	1	.0001	.999
2, 3	Filter	2	.002	
4, 5, 6, 7	Regulator	4	.160	
8, 9	Pressurizing Valve	2	.050	
10	Shut-off Valve	1	.198	
11	Tank, Oxidizer	1	.220	
12	Tank, Fuel	1	.220	
13, 14, 15, 16	Relief Valve	4	.016	
17, 18, 19, 20	Solenoid Valve	4	.050	
21, 22, 23, 24	Check Valve	4	.020	
25, 26	Level Detector	2	.150	
27, 28	Isolation Valve	2	.020	
29, 30, 31, 32	Valve Control*	4	.125	
33, 34, 35, 36	Filter	4	.002	
37	Thrust Chamber	1	.032	
38, 39, 40, 41	Squib	4	.0001	
42, 43, 44	Solenoid Valve	3	.050	
45, 46, 47	Thrust Jets	3	.001	

* Valve Control consists of one solenoid valve, one actuator, and two check valves.

4. A command signal can actuate the shut-off valves, sealing off the solenoid valve. This will degrade attitude capability in one axis.

(d) Mathematical Model

The mathematical model is taken from the cold gas system schematic shown in Figure 4.5.2-5.

TABLE 4.5.2-7. ORBITER COLD GAS SYSTEM

Component No.	Component	Quan.	Each Component: Failure Rate %/1000 hrs	Each Component: Reliability
1, 2	Tank, Gas	2	.0001	.99
3, 4	Filter	2	.002	
5, 6, 7, 8	Pressure Regulator	4	.160	
9 — 14	Shut-off Valve	6	.198	
15 — 26	Solenoid Valve	12	.050	
27	Latch Valve	1	.050	

$$R_{(\text{cold gas system})} = R^2_{(\text{tanks})} \cdot R^2_{(\text{filter})} \cdot \left[1 - \left(1 - R_{(\text{pressure regulator})}{}^2\right)\right]^2 \cdot R^6_{\substack{(\text{shut-off})\\ \text{valves}}} \cdot R^{12}_{\substack{(\text{solenoid})\\ \text{valves}}} \cdot R_{(\text{latch valve})}$$

Sensing or fill components are not included in the mathematical model.

Substituting the value previously defined for the cold gas system:

$$R_{(\text{cold gas system})} = .99$$

4.5.3 LANDER SYSTEM

A. System Definition

The Lander System comprises two Lander vehicles which are essentially identical. Each is independent of, and in working redundancy with the other.

The function of the Lander system is to monitor Martian atmospheric and surface conditions and to perform specified scientific experiments during the entry, descent, and surface phases of the Lander mission. In addition, the acquired data must be recorded and periodically communicated to Earth.

The Voyager System design provides two modes of Lander communication; i.e., the relay mode (Lander to Orbiter to Earth) and the direct mode (Lander to Earth). During the first three months of the Lander surface phase, the direct mode of communication is a backup for the relay mode. Thereafter (through the 6th month of the surface phase) the Landers will monitor seasonal changes in atmospheric and surface conditions, and will utilize the direct mode to communicate the data to Earth.

(1) Reliability Analysis

This section presents the final reliability estimate analysis which was performed for the Mars 1969 Lander. Reliability estimates for the design were performed continually throughout the study phase to assess the current reliability design status and to evaluate the effect of design changes on the Lander system reliability. Considerable reliability growth has been achieved during the course of the study, such that the numerical estimate has increased from approximately 23 percent to 93.7 percent. This substantial increase in estimated Lander reliability was due primarily to the following factors:

1. **Initial parts counts for items such as the Lander communications, power controllers and power regulators were considerably revised reflecting later design simplifications and actual circuit schematics and circuit block diagrams.**

2. **The programmer and data multiplexer incorporate the extensive use of majority logic redundancy.**

3. **Additional working and standby redundancy were incorporated in the thermal control design and in various other subsystem applications.**

4. **Operating times for many components were significantly reduced by duty cycling and time sharing rather than continuous operation.**

5. **Consideration primarily of those failure modes of devices which contribute to system failure; many failures contribute only to causing system degradation (less than 100 percent system effectiveness).**

The Lander vehicle reliability and subsystem reliabilities for each mission phase are summarized in Table 4.5.3-1. The percentage of total system failure rate is given for each subsystem for each phase of the mission in order to assess the relative magnitude of risk involved and to highlight those areas where reliability improvement effort should be concentrated.

A detailed analysis of each individual Lander subsystem is given in the following sections. Each subsystem analysis includes the following:

1. Description of subsystem and reliability features

2. Mathematical model and reliability computation

3. Block diagram.

(2) Mathematical Model and Reliability Computation

The Lander vehicle design has been subdivided into six functional subsystems. As stated above, the reliability analysis of each subsystem is treated separately in later sections of this study report. The quantitative reliability estimates for each of the subsystems are entered in the Lander System Mathematical Model to obtain the estimated reliability of the Lander system.

$$R_{(\text{Lander System})} = 1 - \left(1 - R_{(\text{Lander})}\right)^2$$

Where:

$$R_{(\text{Lander})} = R_{(\text{Communications})} \cdot R_{(\text{EP\&D})}$$

TABLE 4.5.3-1. LANDER VEHICLE RELIABILITY SUMMARY (6 MONTHS SURFACE MISSION)

Lander Vehicle Subsystems	Transit and Entry t 7000 Hours	% Contribution	Surface t ≅ 4400 Hours	% Contribution	Total Mission	% Contribtuion
Communications	--	--	.921	55.1	.921	28.4
Elect Pwr & Dist.	.963	26.8	.976	16.3	.940	21.4
Propulsion & Sep.	.968	23.0	--	--	.968	11.1
Thermal Control	.945	40.2	.977	15.7	.923	27.6
Retardation	.986	10.0	--	--	.986	4.9
Orientation	--	--	.981	12.9	.981	6.6
Total	.869	100.0	.861	100.0	.748	100.0

$$\cdot R_{(\text{Propulsion \& Separation})} \cdot R_{(\text{Thermal Control})}$$

$$\cdot R_{(\text{Retardation})} \cdot R_{(\text{Orientation})}$$

Using the Lander subsystem reliability values tabulated in Table 4.5.3-1 gives:

$$R_{(\text{Lander})} = (.921)\ (.940)\ (.968)\ (.923)\ (.986)\ (.981)$$

$$R_{(\text{Lander})} = .748$$

Entering the above value in the equation for the reliability of the Lander system gives:

$$R_{(\text{Lander System})} = 1 - (1 - .748)^2 = 1 - (.063)$$

$$R_{(\text{Lander System})} = .937$$

Therefore, the probability of at least one Lander vehicle successfully completing a six month mission on the surface of Mars is 93.7 percent.

Lander system and subsystem reliability values also were computed for 100 hour and 3 month surface missions. Table 4.5.3-2 summarizes Lander system, vehicle, and subsystem reliability values for all 3 mission periods.

TABLE 4.5.3-2. SUMMARY OF RELIABILITY VALUES

Lander Vehicle Subsystems	Reliability (R)		
	100 Hours	3 Months	6 Months
Communications	.998	.962	.921
Electrical Power & Distribution	.963	.951	.940
Propulsion & Separation	.968	.968	.968
Thermal Control	.945	.932	.923
Retardation	.986	.986	.986
Orientation	.981	.981	.981
Lander Vehicle Reliability (π R)	.850	.798	.748
Lander System Reliability (At least one Lander vehicle surviving)	.978	.959	.937

B. Subsystem Definition

(1) Communications Subsystem

(a) Reliability Analysis

The PLM/FM Communications subsystem provides the functions of Telemetry, Tracking, and Command. Increased probability of operation is provided by two separate communication links; a prime VHF(100 mc) link via Orbiter relay and a reduced data rate backup link which communicates directly with the Earth at SHF (2.2 kmc). All cummunications equipment will be primarily solid state using the latest available thin film, high density packaging techniques with the exception of a few selected experiments and the SHF klystrons. This subsystem will be designed for gradual degradation rather than total catastrophic failure by judicious use of majority logic redundancy, both in the command and telemetry portions.

(b) Mathematical Model and Reliability Computation

The mathematical model given below for the communications subsystem is based on the subsystem block diagram (Figure 4.5.3-1) and the data given in Table 4.5.3-3. In addition, it reflects the alternate mode, standby redundancy which exists in the communications link during the first three months of the surface mission. The subscripts in the equation refer to the identification numbers assigned to the subsystem components (Reference Table 4.5.3-3), and to the mission time (in hours).

$$R_{\substack{\text{(Comm.}\\ \text{Subsystem)}}} = R_{\substack{1-5\\(4400)}} \left[R_{\substack{6-12\\(2200)}} + \frac{\lambda_{6-12}}{\lambda_{13-23} - \lambda_{6-12}} \left(R_{\substack{6-12\\(2200)}} - R_{\substack{13-23\\(2200)}} \right) \right] R_{\substack{13-23\\(2200)}}$$

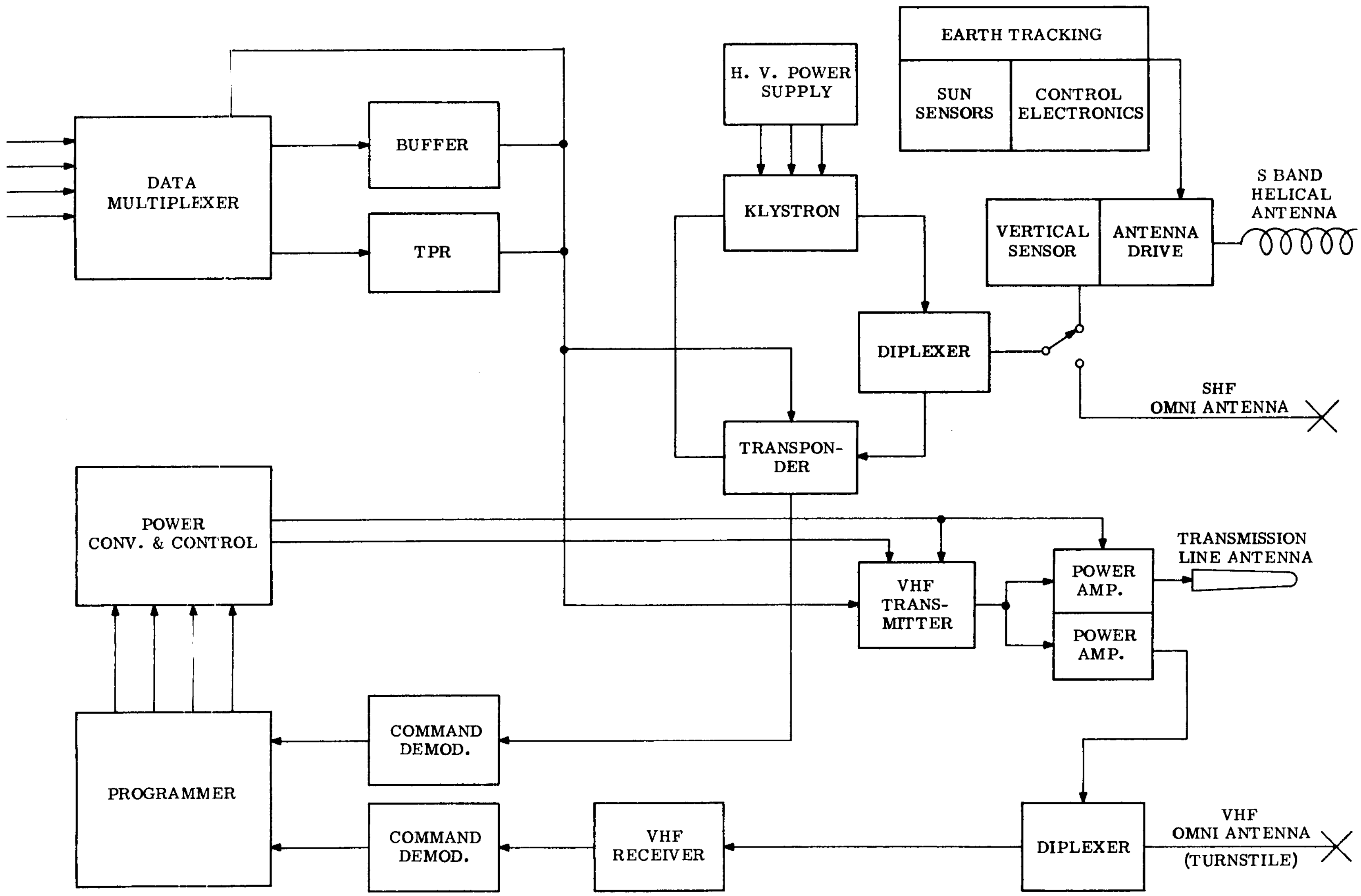


Figure 4.5.3-1. Simplified Block Diagram Lander Communication Subsystem

TABLE 4.5.3-3. LANDER COMMUNICATIONS SUBSYSTEM RELIABILITY DATA

Communications Subsystem Components	Failure Rate λ(%/1000 Hrs.)	t	% Duty Cycle	Reliability R
1. Programmer	.698	4400	100	.970
2. Data Multiplexer	2.900		16.7	.980
3. Power Supply	.323		16.7	.998
4. Thermoplastic Recorder	3.18		16.7	.977
5. Buffer (100 KB Recorder	3.50	≈1 Hr.	100	≈ 1.000
	Σ λ = 3.921			πR = .927
6. VHF Transmitter	.088	2200	3.7	.9999
7. Power Amplifiers	.089			.9999
8. Diplexer	.019			≈ 1.0000
9. VHF Receiver	.161			.9999
10. Command Demodulator	.254			.9998
11. Turnstile Antenna	-			-
12. Transmission Line Antenna	-	≈1 Hr.	100	-
	Σ λ = 0.611			πR = .9995
13. Klystron	1.000	4400	3.7	.9984
14. Power Supply, H.V.	.249			.9996
15. Diplexer	.019			.9999
16. Transponder	2.570			.9958
17. Command Demodulator	.254			.9999
18. Helical Antenna				
19. Omni Antenna				
20. Vertical Sensor	3.94			.9936
21. Sun Sensor				
22. Antenna Drive				
23. Control Electronics				
	Σ λ = 8.032			πR = .987 (4400 Hrs.) .9935 (2200 Hrs.)

Entering the reliability values from Table 4.5.3-3 gives:

$$R_{(\text{Comm. Subsystem})} = (.927)\left[.9995 + \frac{.611}{8.032-.611}(.9995 - .9935)\right](.9935)$$

$$R_{(\text{Comm. Subsystem})} = .921 \text{ (6 months surface mission)}$$

The mathematical model given above for the six months surface mission must be modified for the three month and the 100 hour missions. For the latter two cases, the mission times are reduced, and the direct mode of communication (components 13 through 23 of Table 4.5.3-3) serves only as an alternate to the relay mode (components 6 through 12). The modified mathematical model for the reduced mission times is as follows:

$$R_{\text{(Comm. Subsystem)}} = R_{1-5}\left[R_{6-12} + \frac{\lambda_{6-12}}{\lambda_{13-23} - \lambda_{6-12}}\left(R_{6-12} - R_{13-23}\right)\right]$$

The component reliability values given in Table 4.5.3-3 were modified to reflect the changes in mission time, and were entered in the mathematical model above to obtain the communication subsystem reliability values given below.

$$R_{\text{(Comm. Subsystem)}} = .998 \text{ (100 hour surface mission)}$$

(2) Electrical Power and Distribution

(a) Reliability Analysis

Generation of electrical power for the Lander system is provided by means of the Radioisotopic Thermoelectric Generator supplemented by rechargeable nickle-cadium batteries during peak power periods. An additional function of the RTG is to provide a source of heat used for Lander thermal control. Power control is accomplished by switching functions initiated by the command portion of the communications system. Distribution will be provided by cabling harnesses to individual subsystems and components.

(b) Mathematical Model and Reliability Computation

The mathematical model given below for the Electrical Power and Distribution subsystem is based on the subsystem block diagram (Figure 4.5.3-2) and the data given in Table 4.5.3-4. The subscripts to each of the "R" factors refer to the identification numbers assigned to each of the subsystem components in Table 4.5.3-4.

$$R_{\text{(EP\&D)}} = \left[R_1\ R_2\ R_3\ R_4\ R_{5\text{(Transit \& Entry)}}\right]\left[R_1\ R_2\ R_3\ R_4\ R_{5\text{(Surface)}}\right]$$

Entering the component reliability values tabulated in Table 4.5.3-4 gives the estimated reliability of the Electrical Power and Distribution subsystem for a six month surface mission.

$$R_{\text{(EP\&D)}} = (.998)\ (.985)\ (.996)\ (.993)\ (.990)\ (.999)\ (.991)\ (.998)\ (.996)\ (.992)$$

$$R_{\text{(EP\&D)}} = .940 \text{ (6 month surface mission)}$$

The mathematical model given above for a six month surface mission applies also to the 100 hour and the 3 month surface missions. In each of the latter two cases, the component reliability values remain the same for the transit and entry phase, but the values for the surface phase are modified by the decrease in surface mission time. The estimated reliability of the EP&D subsystem for the 100 hour and the three month mission were calculated on this basis and are given below.

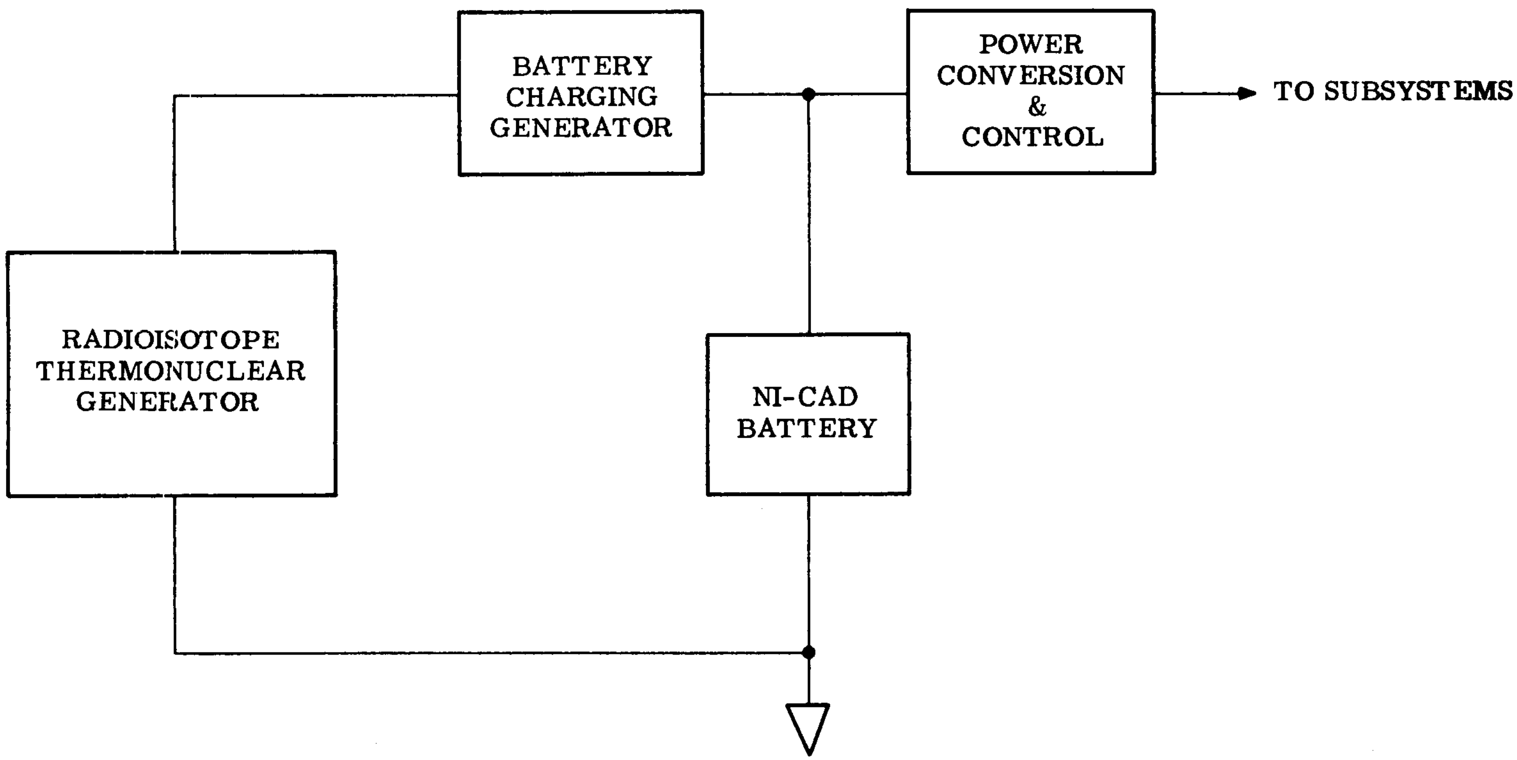


Figure 4.5.3-2. Simplified Block Diagram- Lander Electrical Power and Distribution Subsystem

TABLE 4.5.3-4. LANDER ELECTRICAL POWER AND DISTRIBUTION SUBSYSTEM RELIABILITY DATA

Electrical Power and Distribution Subsystem Components	Quan.	Failure Rate (%/1000 hrs)	Transit + Entry		Surface		Reliability	
			% Duty Cycle	hrs	% Duty Cycle	hrs	Transit + Entry	Surface
1. Radioisotopic-Thermoelectric Generator	1	.028	100	7000	100	4400	.998	.999
2. Battery Charging Regulator	1	.211	100	7000	100	4400	.985	.991
3. NI-CAD Battery	1	.050	100	7000	100	4400	.996	.998
4. Harness, Cabling, Connectors	—	.100	100	7000	100	4400	.993	.996
5. Power Conversion and Control	1	.147 transit .175 surface	100	7000	100	4400	.990	.992

$R_{(EP\&D)}$ = .951 (3 month surface mission)

$R_{(EP\&D)}$ = .963 (100 hour surface mission)

(3) Propulsion and Separation

(a) Reliability Analysis

This subsystem provides separation from the Orbiter, spin stabilization, and transfer into the planetary entry trajectory. Past experience with previous successful GE/MSD programs will be applied to produce the most reliable configuration possible. Initial mechanical and electrical separation will be effected by explosive bolts and in-flight disconnects (each with redundant squibs). Subsequent separation and spin stabilization will be performed by a cold gas system and trajectory insertion by means of a solid rocket motor. All commands will be pre-programmed into the Lander programmer and power will be supplied by the peaking batteries.

(b) Mathematical Model and Reliability Computation

The mathematical model given below for the Propulsion and Separation subsystem is based on the subsystem block diagram (Figure 4.5.3-3) and the data given in Table 4.5.3-5. The subscripts to each of the "R" factors refer to the identification numbers assigned to each of the subsystem components in Table 4.5.3-5. Where redundancy exists within a component, it has been considered in calculating the "R" value for that component.

$$R_{(\text{Prop \& Sep})} = R_1 \cdot R_2 \cdot R_3 \cdot R_4 \cdot R_5 \cdot R_6 \cdot R_7 \cdot R_8 \cdot R_9 \cdot R_{10}$$

Entering the component reliability values tabulated in Table 4.5.3-5 gives the estimated reliability of the Propulsion and Separation subsystem. The reliability of the subsystem is not affected by the duration of the surface mission on Mars.

$$R_{(\text{Prop \& Sep})} = (.999)\,(.996)\,(.9994)\,(.9911)\,(.9986)\,(.9994)\,(.9911)\,(.9986)\,(.999)\,(.996)$$

$$R_{(\text{Prop \& Sep})} = .968$$

(4) Thermal Control Subsystem

(a) Reliability Analysis

This subsystem provides active thermal control for the Lander. The prime purpose of the subsystem is to dissipate excess heat generated by the RTG. This is accomplished by convection and thermal radiation during the in-transit and surface phases, and by liquid evaporation during boost and entry. A portion of the excess heat is utilized to maintain the temperature of internal components within specified design limits.

Working and standby redundancy are used extensively to reduce the probability of failure of the subsystem.

(b) Mathematical Model and Reliability Computation

The mathematical model given below for the Thermal Control subsystem is based on the subsystem block diagram (Figure 4.5.3-4) and the data given in Table 4.5.3-6. The transit and surface phases are treated separately (for simplicity), and the redundancy applicable to each phase is included. The subscripts used refer to the identification numbers assigned to the subsystem components in Table 4.5.3-6.

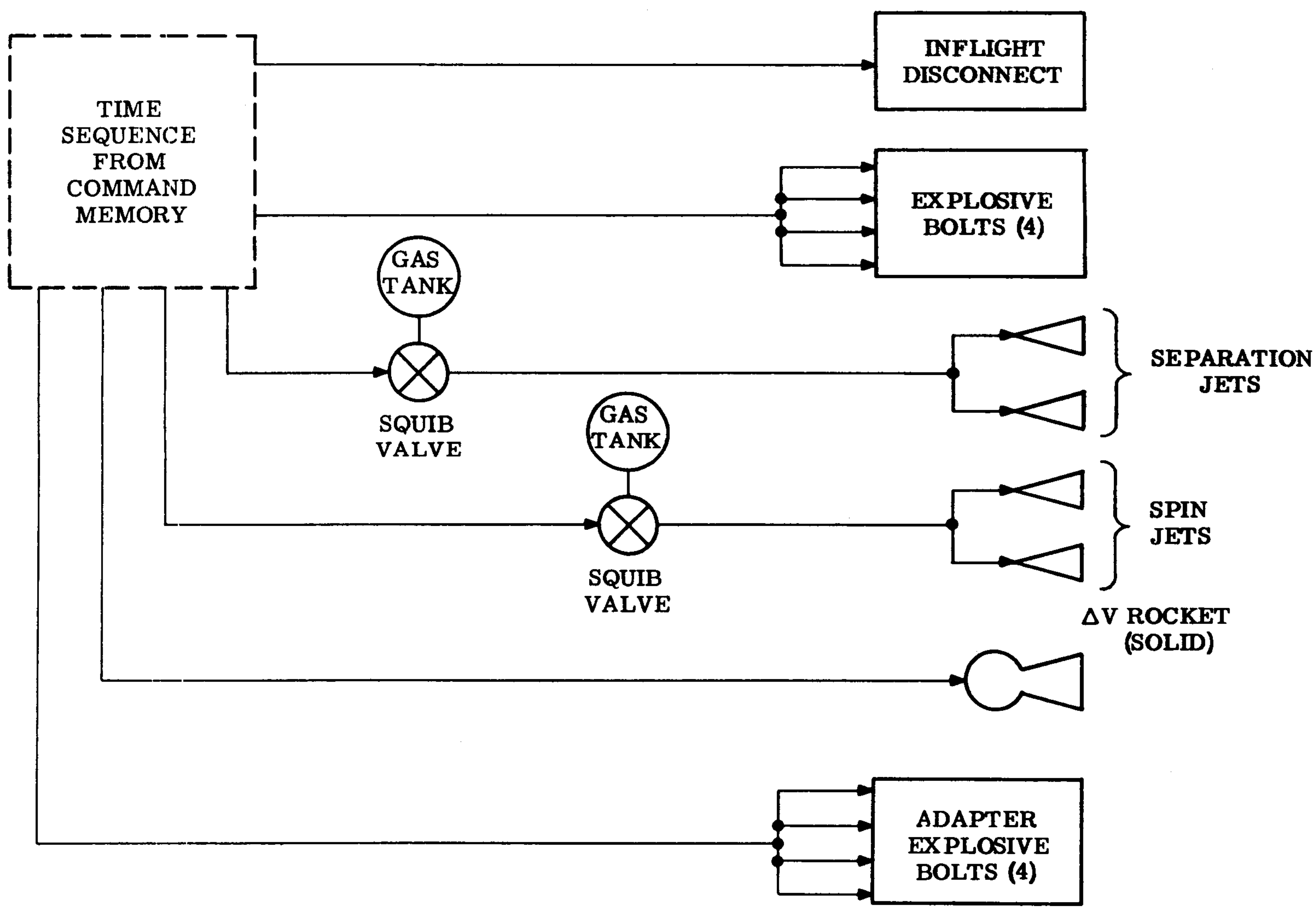


Figure 4.5.3-3. Simplified Block Diagram- Lander Propulsion and Separation Subsystem

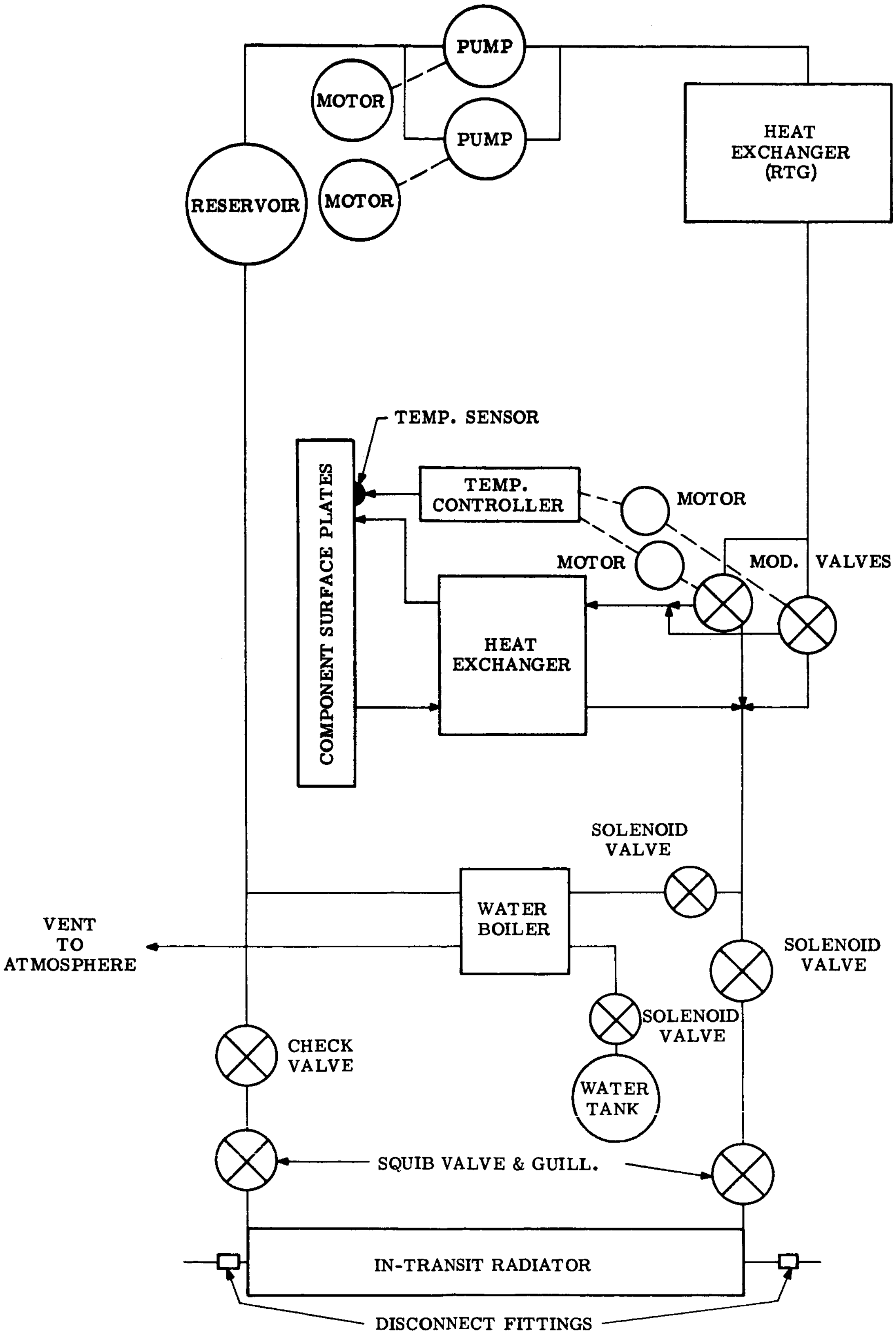


Figure 4.5.3-4. Thermal Control Subsystem Block Diagram

TABLE 4.5.3-5. LANDER PROPULSION AND SEPARATION SUBSYSTEM RELIABILITY DATA

Propulsion and Separation Subsystems Components	Quan.	Leakage F.R. $\lambda\ 10^{-5}$ Failures/Hrs.	Operation F.R. $\lambda\ 10^{-3}$ Failures/Oper.	Transit Hours	Reliability	Remarks
1. Inflight Disconnect, Orbiter	1	—	1	7000	.999	Red. Squibs
2. Orbiter Explosive Bolts	4	—	1	7000	.996	Red. Squibs
3. Gas Tank (Separation)	1	.008	—	7000	.9994	AVCO F.R. for Squib/ solenoid valves = .227%/1000 hours
4. Squib Valve (Separation)	1	.113	1	7000	.9911	
5. Jets and Plumbing (Separation)	2	.010	—	7000	.9986	
6. Gas Tank (Spin)	1	.008	—	7000	.9994	Assume, leakage = .113%/ 1000 hours, prob. of premature = 0
7. Squib Valve (Spin)	1	.113	1	7000	.9911	
8. Jets and Plumbing (Spin)	2	.010	—	7000	.9986	
9. Delta-V Solid Rocket	1	—	1	7000	.999	Red. Squibs
10. Adapter Explosive Bolts	4	—	1	7000	.996	Red. Squibs
					π = .968	

TABLE 4.5.3-6. LANDER THEMAL CONTROL SUBSYSTEM RELIABILITY DATA

Thermal Control Subsystem Components	Quan.	Failure Rate $\lambda . 10^{-5}$ (Failures/hr)	Transit and Entry		Surface		Reliability		Remarks
			% Duty Cycle	Hrs	% Duty Cycle	Hrs	Transit & Entry	Surface	
1. Water Tank	1	.035	100	7000	—	—	.9976	—	Not used on surface
2. Water Boiler	1	.110			—	—	.9923	—	Not used on surface
3. Solenoid Valve	1	.113			—	—	.9921	—	Not used on surface
4. RTG Heat Exchanger	1	.110			100	4400	.9923	.9952	
5. Liq. to Liq. Heat Exchanger	1	.110					.9923	.9952	
6. Pumps	2	.112					.9922	.9951	Standby redundancy
7. DC Motors	2	.112					.9922	.9951	
8. Solenoid Valve	1	.113	—	—			—	—	Intransit Premature negligible Not used on surface
9. Solenoid Valve	1	.113	—	—			—	.9950 Redundant on surface	Premature negligible
10. Squib Valve and Buillotine	2	.113	—	—			—	.9950	Premature negligible
11. Check Valve	1	.011	—	—			—	.9995	No effect in transit
12. In-Transit Radiator	1	.110	100	7000	—	—	.9923	.9952	
13. Accumulator (Reservoir)	1	.035			100	4400	.9976	.9985	
14. Modulation Valves	2	.145					.9900	.9936	Standby redundancy
15. DC Motors	2	.112	1		10		.9922	.9951	
16. Temp. Sensor	1	.015	100		100		.9990	.9993	
17. Temp. Controller	1	.047					.9967	.9979	
18. Plumbing, Fittings, Tubing and Component Surface Plates	—	.110					.9923	.9952	Valve operations when required assumed leakage F.R. = 1/2 generic F.R. for valves assumed heat exchanger F.R. = 1/2 generic AVCO F.R.

$$R_{(\text{Transit Phase})} = R_1 \cdot R_2 \cdot R_3 \cdot R_4 \cdot R_5 \cdot R_6 \cdot R_7 \cdot \left[1 + (\lambda_6 + \lambda_7)t\right] \cdot R_{12} \cdot$$
$$R_{13} \cdot R_{14} \cdot R_{15} \cdot \left[1 + (\lambda_{14} + \lambda_{15})\, t\right] \cdot R_{16} \cdot R_{17} \cdot R_{18}$$

$$R_{(\text{Surface Phase})} = R_4 \cdot R_5 \cdot R_6 \cdot R_7 \cdot \left[1 + (\lambda_6 + \lambda_7)\, t\right] \cdot \left[1 - (1-R_9)(1-R_{10})\right] \cdot$$
$$\left[1 - (1-R_{10})(1-R_{11})\right] \cdot R_{12} \cdot R_{13} \cdot R_{14} \cdot R_{15}$$
$$\left[1 + (\lambda_{14} + \lambda_{15})\, t\right] \cdot R_{16} \cdot R_{17} \cdot R_{18}$$

$$R_{(\text{Thermal Control})} = R_{(\text{Transit Phase})} \cdot R_{(\text{Surface Phase})}$$

Entering the "R" and "λ" values from Table 4.5.3-6 gives:

$$R_{(\text{Thermal Control})} = (.9454)\,(.9766) = .9233 \text{ (6 Month Mission)}$$

The Thermal Control subsystem mathematical model given above for the 6 month surface mission is valid also for the 3 month and the 100 hour missions. The reliability value for the transit phase is the same for all mission periods, but that for the surface phase is modified by the change in surface mission time. The reliability values for the reduced mission times were computed on this basis and are as given below.

$$R_{(\text{Thermal Control})} = .932 \text{ (3 months surface mission)}$$

$$R_{(\text{Thermal Control})} = .945 \text{ (100 hours surface mission)}$$

(5) Retardation Subsystem

(a) Reliability Analysis

This subsystem will retard the Lander vehicle during atmospheric entry to provide time for experimentation during descent and to minimuze landing impact. Retardation will be performed by means of a deceleration parachute and a main parachute. Landing impact will be absorbed by the structural honeycomb crush-up material. As in the orientation subsystem, the retardation design must accommodate a wide range of environmental conditions due to trajectory uncertainty at the time of entry and the unknown Mars atmosphere. Experience based on past GE/MSD programs will be applied on this program to provide the most reliable design over a wide range of environments. Redundant programming and trajectory sensing as well as redundant initiation of pyrotechnics will be used. This subsystem, by necessity, will be completely independent of other subsystems with respect to programming and power requirements in order to assure successful entry and landing.

(b) Mathematical Model and Reliability Computation

The mathematical model for the Retardation subsystem is given below. It is based on the subsystem block diagram (Figure 4.5.3-5) and on the data given in Table 4.5.3-7. Where redundancy exists within a component, it has been included in the computation of the "R" value for that component. The subscripts used in the mathematical model refer to the identification numbers assigned to each of the subsystem components in Table 4.5.3-7.

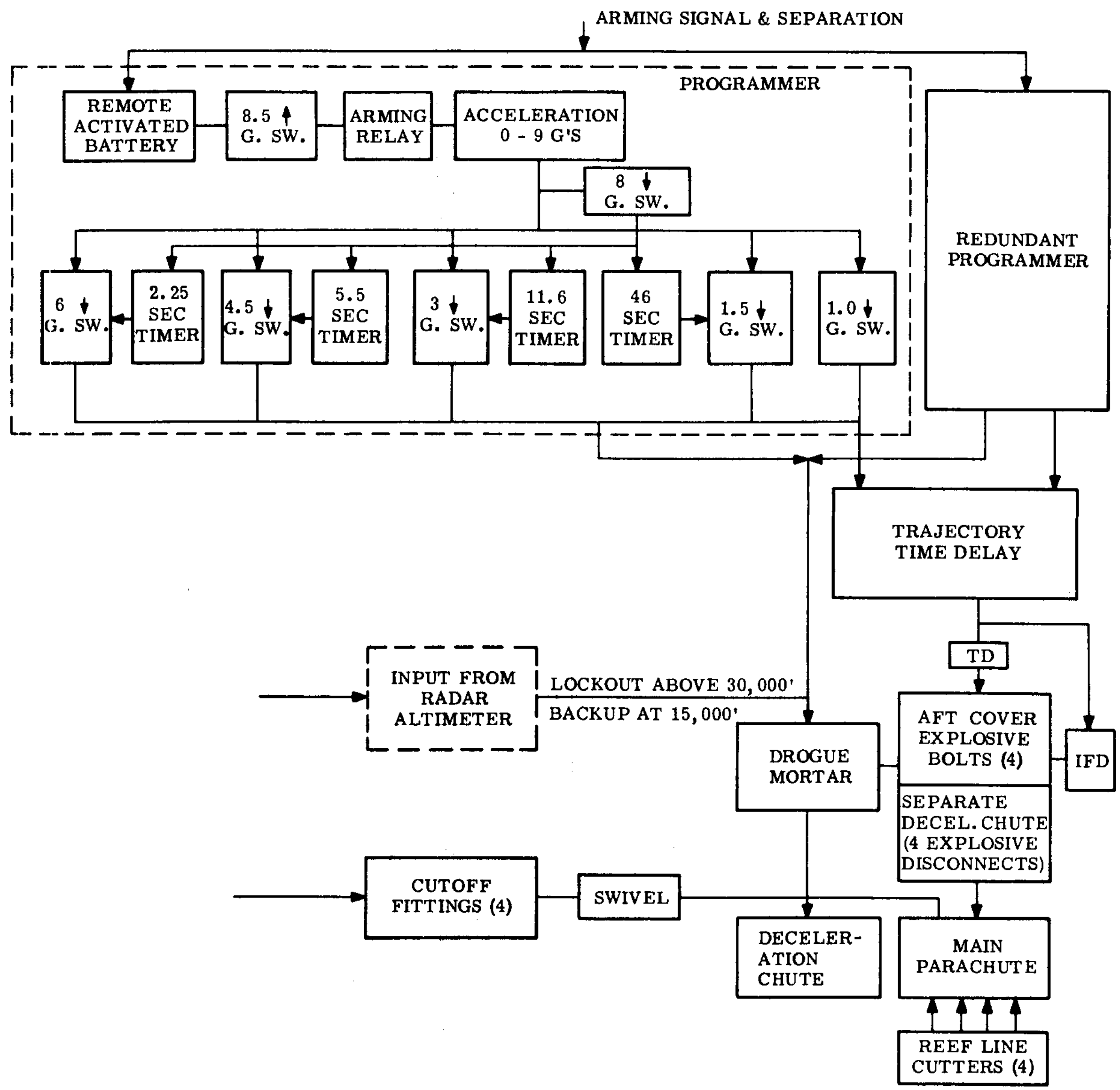


Figure 4.5.3-5. Block Diagram- Lander Retardation Subsystem

TABLE 4.5.3-7. LANDER RETARDATION SUBSYSTEM RELIABILITY DATA

Retardation Subsystem Components	Quan.	Operation F.R. $\lambda 10^{-3}$ Failures/Mission	Reliability	Remarks
1. Remote Activated Batteries	2	2	.998	(2 Redundant Squibs)
2. Arming Relay	2	<.1	>.9999	
3. G Switches	14	2	.998	Redundant
4. Timers	8	1	.999	Programmers
5. Time Delay, Trajectory	1	<.1	>.9999	
6. Drogue Mortar	1	1	.999	(2 Red. Squibs)
7. Decel. Chute	1	1	.999	
8. Inflight Disconnect	1	1	.999	(2 Red. Squibs)
9. Time Delay	1	<.1	>.9999	
10. Aft Cover Explosive Bolts	4	1	.999	(2 Red. Squibs)
11. Decel. Chute Explosive Disconnects	4	1	.999	(2 Red. Squibs)
12. Main Parachute	1	1	.999	
13. Swivel	1	<.1	>.9999	
14. Reef Line Cutters	4	<.1	>.9999	One of four required
15. Cutoff Fittings	4	<.1	>.9999	(2 Red. Squibs)

$$R_{(Retardation)} = \left[1 - (1-R_1 R_2 R_3^7 R_4^4)^2\right] R_5 \cdot R_6 \cdot R_7 \cdot R_8 \cdot R_9 \cdot R_{10}^4 \cdot R_{11}^4 \cdot$$

$$R_{12} \cdot R_{13} \cdot \left[1 - (1-R_{14})^4\right] \cdot R_{15}$$

Entering the "R" values given in Table 4.5.3-7 gives:

$$R_{(Retardation)} = .986$$

The reliability of the Lander Retardation subsystem is not affected by the duration of the Lander surface mission.

(6) Orientation Subsystem

(a) Reliability Analysis

Orientation of the Lander vehicle on the surface of Mars, including the deployment of experiments, is performed by the orientation subsystem. The selection of the final design configuration of side orientation was based on the minimum number of functions and operations required to orient. The major problem in the subsystem design was to accommodate the range of surface terrain conditions which could be expected and initial Lander orientation after impact. The sequence for orientation is pre-programmed in the command programmer and will repeat until orientation is achieved, barring extreme circumstances.

(b) Mathematical Model and Reliability Computation

The mathematical model given below for the orientation subsystem is based on the subsystem block diagram (Figure 4.5.3-6) and the data given in Table 4.5.3-8. The subscripts to each of the "R" factors refer to the identification numbers assigned to each of the subsystem components in Table 4.5.3-8. Each component "R" value given in Table 4.5.3-8 has been calculated for the total required quantity of that component.

$$R_{(Orientation)} = R_1 \cdot R_2 \cdot R_3 \cdot R_4 \cdot R_5 \cdot R_6 \cdot R_7 \cdot R_8 \cdot R_9 \cdot R_{10} \cdot R_{11} \cdot R_{12} \cdot R_{13}$$

Entering the "R" values from Table 4.5.3-8 gives:

$$R_{(Orientation)} = .990$$

The probability of surviving impact and encountering suitable terrain has been estimated to be .990. The reliability of the subsystem has beem modified by this factor.

$$R_{(Orientation)} = (.990)\,(.990) = .981$$

The reliability of the Orientation subsystem is not affected by the duration of the surface mission on Mars.

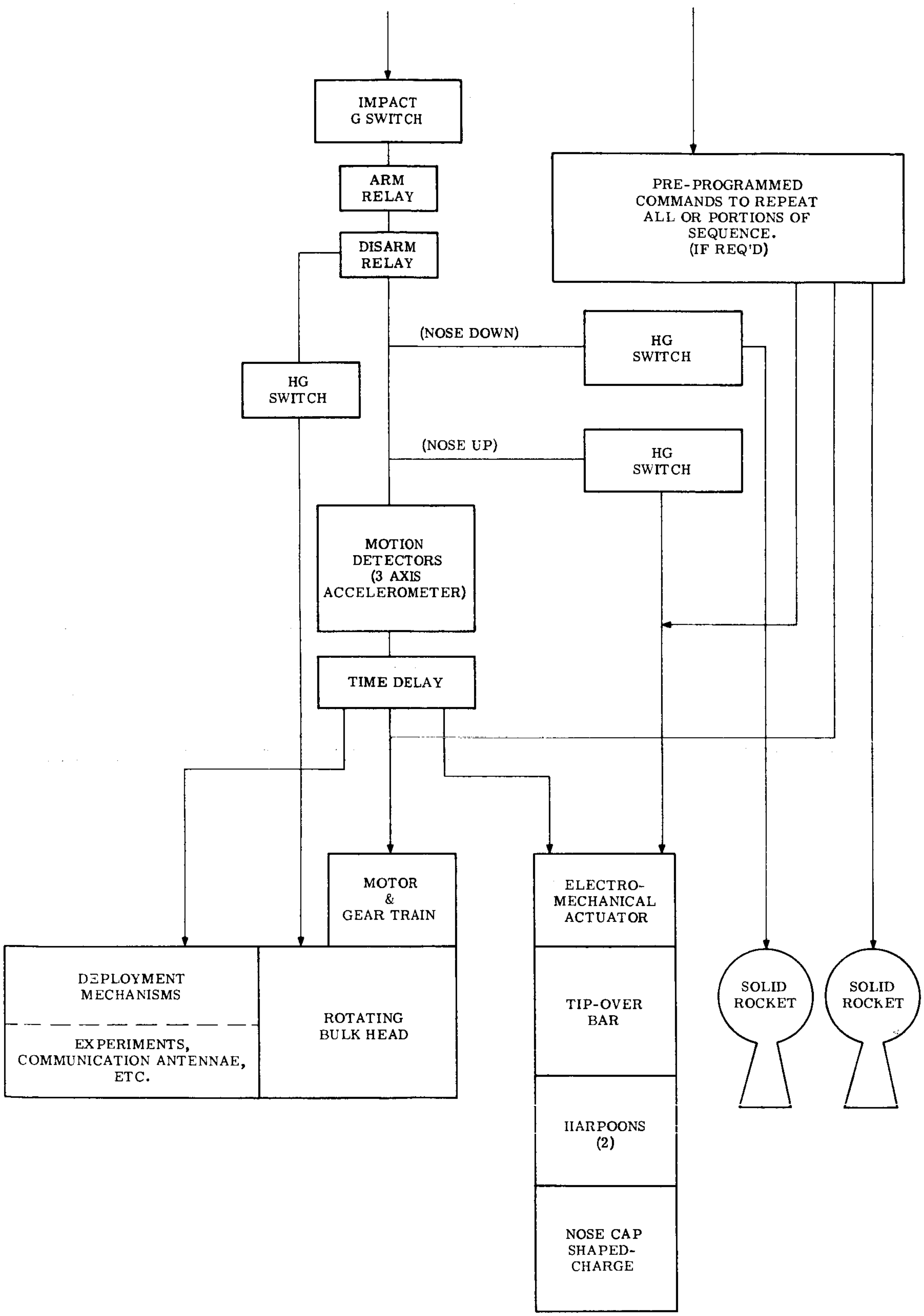


Figure 4.5.3-6. Block Diagram- Lander Orientation Subsystem

TABLE 4.5.3-8. LANDER ORIENTATION SUBSYSTEM RELIABILITY DATA

Orientation Subsystem Components	Quan.	Operation F. R. $\lambda\ 10^{-3}$ Failures/Mission	$\Sigma N \lambda M$	R Reliability	Remarks
1. G Switch, Impact	1	1	.001	.999	
2. Arm Relay	1	< .1	< .0001	> .9999	
3. Disarm Relay	1	< .1	< .0001	> .9999	
4. Mercury Switches	3	< .1	< .0001	> .9999	
5. Motion Detectors	3	.33	.001	.999	
6. Time Delay	1	< .1	< .0001	> .9999	
7. Motor and Gear Train	1	1	.001	.999	
8. Deployment Mechanisms	≈ 10	.1	.001	.999	Initiation by Red. Squibs
9. Electro-Mechanical Actuator	1	1	.001	.999	Initiation by Red. Squibs
10. Tilt Bar	1	< .1	< .0001	> .9999	Initiation by Red. Squibs
11. Harpoons	2	1	.002	.998	Initiation by Red. Squibs
12. Solid Rockets	2	1	.002	.998	Initiation by Red. Squibs
13. Shaped Charge	1	1	.001	.999	Initiation by Red. Squibs

4.6 IMPLEMENTATION

A summary of Reliability and Quality Assurance recommendations and conclusions has been prepared as noted under 4.2.2 above.

4.6.1 REQUIREMENTS

As an approach to their implementation, an initial draft of "Reliability Requirements for Voyager Contractors and Subcontractors" has been prepared as GE document S-31100. It is sufficiently comprehensive to establish requirements at all contractual levels and it has been written with the intent that the specification itself or its provisions will be suitable for use in or for reference by NASA contractual documents as well as by those of the contractors and subcontractors. Additional copies of the GE Specification are available to NASA upon request.

It would be intended that this document, S-31100 (or its equivalent), be maintained, revised, and improved by the Voyager Contractor, reviewed and approved by NASA, and applied during the preliminary design period as well as during the equipment development and production periods of the Voyager Program.

The systems reliability assessments incorporated into S-31100, section 6.2 have been reviewed as final adjustments of configuration were determined and also as input documents and presentations were received from other participating companies. These Reliability Requirements remain essentially unchanged and are considered directly applicable as requirements for achieving the three successes out of four opportunities recommended as the basic Voyager System Reliability Requirement.

4.6.2 REVISIONS OF REQUIREMENTS FOR LATER OPPORTUNITIES

Attention has been given to extending the parametric summary of the basis and criteria for the recommendation of dual Landers for Mars 1969 to include the full range of Lander sizes and scientific instruments which might be considered for these and other Voyager launch opportunities. It is recognized that the objectives as well as the instrumentation of later missions may be expected to be altered extensively as a result of information obtained from prior missions (e.g., Mariner B). Were life to be detected, its classification and study would immediately dominate instrument selection and application. Roving vehicles may logically be applied to later flights. However, the review of the instruments applicable to the presently planned Voyager missions has confirmed the allocation of percent relative Mission Value given to these instruments as being sufficiently representative of any selection which might presently be made.